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U.S. AIR FORCE
Project RAND

PROJECT FEED BACK
SUMMARY REPORT

March 1, 1954

R-262

Volume II

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PROJECT RAND

Contract No. AF 33(038)-6413

PROJECT FEED BACK

SUMMARY REPORT

Edited by

J. E. LIPP and R. M. SALTER

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CONTENTS

B-1

VOLUME II

This volume contains information on the specific design aspects of the Feed Back device. No attempt has been made here to present a balanced picture. Only those items on which appreciable work has been done, or which are unique to the satellite application, are included. In some cases material has been abstracted from separate research memoranda (listed among the References) and is presented along with briefer discussions of relatively less important items.

These data do not always represent the best or even a completely consistent solution. Extensive use of subcontracted studies, some of which are still in progress, has made complete, continued integration impossible. In addition, future research and development effort will disclose new ideas and new problems requiring study.

However, the information presented should be of value to other investigators and should save them the effort of covering the same ground.

Volume II covers three main topics: The Vehicle, Ground Operations, and Other Satellite Applications.

SYMBOLS

A = aerodynamic reference area

C_D = drag coefficient

$\frac{dC_L}{d\alpha}$ = slope of the lift curve

D/m = drag over mass ratio

g = standard acceleration of gravity at sea level

h = altitude (in ft, unless otherwise noted)

I = propellant system specific impulse

I_{alt} = specific impulse at altitude

I_0 = sea-level specific impulse

I_{total} = total impulse, maximum thrust developed times the effective time of burning

n_x = ratio of vehicle axial acceleration to the standard acceleration of gravity at sea level

n_t = ratio of powerplant thrust to initial gross weight

n_n = ratio of vehicle normal acceleration to the standard acceleration of gravity at sea level

O_H = horizontal overlap ratio for television ground-scanning system

O_V = vertical overlap ratio for television ground-scanning system

P_c = rocket motor combustion chamber internal pressure

q = dynamic pressure

R' = local Reynolds number based on an effective temperature T' (see Ref. 7)

t = time (in sec)

t_b = time in seconds of powerplant operation

T/m = thrust over mass ratio

V = velocity (ft/sec)

V_{go} = velocity of satellite relative to ground

W = vehicle gross weight, instantaneous weight

α = aerodynamic angle of attack

ϵ = ratio of rocket motor exit area to throat area

θ = vehicle path angle

ν = ratio of initial usable propellant weight to initial gross weight

ω = structure to gross weight ratio

THE VEHICLE

Pertinent interior design considerations of the vehicle are included in this section, as well as an evaluation of the effects of such exterior phenomena as environment and flight mechanics.

VEHICLE DESIGN

In investigating the vehicle characteristics, it was felt desirable to substitute engineering experience and judgment for many vast parametric investigations. In all cases, attempts were made to adopt the simplest possible systems and to use materials and techniques which are representative of the present state of the art rather than those requiring extensive engineering development. An inherent requirement for reliability underlies many of the choices.

Succeeding parts of this section cover details of specific payload components, environment, flight mechanics, and guidance and control problems.

Payload Packaging Requirements

The general arrangement of basic missile components, such as tanks, main rocket powerplant, etc., is discussed under "Structure and Weight," page 9, and only the various items comprising the payload are discussed here.

In view of the unique packaging requirements of various portions of the payload, the payload components will be discussed in the following order: (1) auxiliary powerplant unit, (2) ascent and orbital guidance, (3) television camera system, (4) data storage and communications system, and (5) climatization.

Weight breakdowns for payload components are given in Table 1, and their relative arrangement is illustrated in Fig. 1.

Auxiliary Powerplant Unit. The auxiliary powerplant, discussed in detail on page 27, is located in the forward portion of the satellite's ogive nose section. The inherent ability of the system to withstand relatively high temperatures eliminates the necessity for climatization,* and the forward location produces

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*Servomotors capable of operating under temperature conditions of 400°F are used for auxiliary drive power.

Table 1

WEIGHT BREAKDOWNS FOR PAYLOAD COMPONENTS*

Component	Weight (lb)	Total Weight (lb)
Auxiliary powerplant		
Water pump	2	
Radiator	55	
Ducting	23	
Turbine	30	500
Generator	25	
Regulator	15	
Increase for 1 kw more	50	
Controls		
Stable platform and yaw gyro	50	
Ascent computer	60	
Horizon scanner	20	280
Acceleration wheels	120	
Control computer	30	
Environment		
Electronic cooling	25	
Containers	35	80
Paraffin	20	
Television and optical system		
Mirrors and lens	50	
Camera mount	20	
Counterbalance	35	270
Camera 1	40	
Camera 2	40	
Common control	85	
Recording system		
Mechanical components	80	
Counterbalance	20	205
Circuits	90	
Programmer	15	
Transmitting and receiving system		
Transmitter and modulator	55	
Receiver	30	
Antenna	30	140
Antenna control	5	
Antenna counterbalance	15	
Switchgear	5	
TOTAL		1475

*Exclusive of hydraulics and electrical motivating equipment.

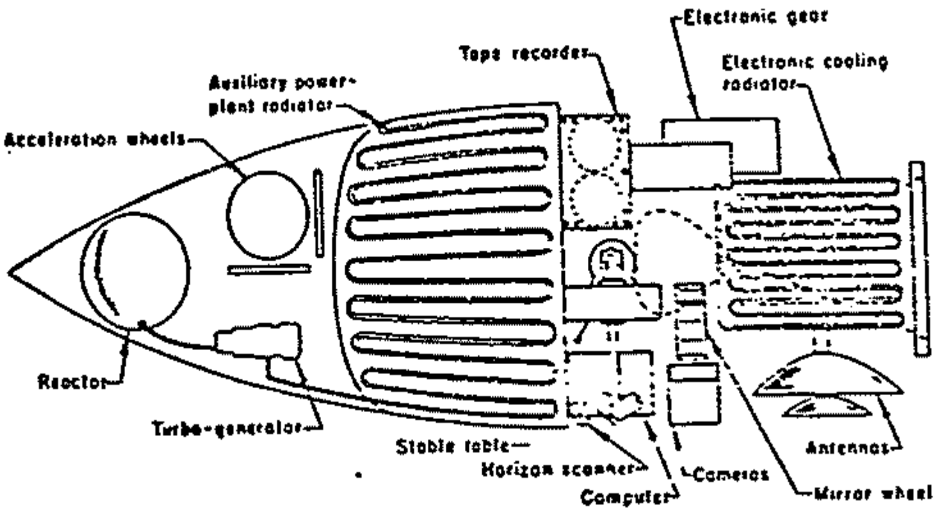

Fig. 1—Schematic of Orbiting Vehicle

a desirable isolation from the electronic equipment. As mentioned elsewhere in this report, heat rejection from the powerplant is obtained through external radiation. The proposed configuration is such that the radiator is located in the outer skin of the fuel and oxidizer tank section* (see Fig. 1). The various heat quantities to be dissipated from the vehicle and the corresponding radiator sizes are summarized in Table 2, below.

Table 2

SUMMARY OF HEAT DISSIPATIONS AND RADIATOR CHARACTERISTICS

Heat Source	System	Radiator Temperature (°F)	Heat Rate (kw)	Radiator Area (ft²)
Powerplant	Mercury	627	74	133
Powerplant	Water (moderator)	300*	4	30
Electronic equipment	Water (cooling)	90	2	56

*68 psia.

Ascent and Orbital Guidance. The major components of the ascent and orbital guidance system include a stable platform, ascent computer, orbital computer, horizon scanner, and acceleration wheels. Acceleration wheels per se have no climatization or shielding requirements and are located just aft of the

*The remaining portion of the vehicle's skin aft of the fuel tank is jettisoned during booster separation.

auxiliary powerplant and immediately forward of the oxidizer tank. The horizon scanner, operating through 360 deg in azimuth at a depression angle of 22 deg, requires vertical stabilization. Since the ascent guidance also requires a stable platform, a single unit common to both systems is used. This platform, along with the horizon scanner, is located in the bottom portion of the satellite vehicle, just aft of the fuel tank, thus providing the unobstructed view necessary for the scanner after the stage separation has occurred. Both of the computers, requiring climatization, are mounted to the port side of the aft bulkhead in the computer box, as shown in Fig. 1.

Television Camera System. Television equipment can be divided into two major groups: (1) the optical unit comprised of the mirror drum with the related lens, mirrors, etc., and (2) the camera unit consisting of Image Orthicon tubes and associated camera control units (see "Television Payload Equipment," page 38). The mirror drum dictates the arrangement of the optical system and the Image Orthicon tubes, since it requires an unobstructed view normal to the longitudinal axis of the missile on the earth side. A good choice is to mount the mirror drum aft of the stable platform and horizon scanner, along with the remainder of the optical system and the Image Orthicon tubes. Camera control units for the Image Orthicons are completely enclosed and are attached to the starboard side of the aft bulkhead.

Data-storage and Communications System. Climatization is needed for the data-storage mechanism (see "Television Payload Equipment," page 38), including both the electronic equipment and the magnetic film. Volume requirements are small, and the entire unit can readily be accommodated in the aft portion of the vehicle, above the main powerplant accessory and turbopump compartment.

Communication equipment consists of the programmer, transmitter and modulator, command receiver, switchgear, and antenna unit. The total volume required for the communication components (excluding the antenna unit) permits the system to be located above the rocket powerplant, along with the data-storage unit. Again the entire unit is climatized.

The antenna unit requires about 8.2 ft^3 to perform the required antenna movement. To provide an unobstructed view of the earth, and at the same time to provide ample volume, the antenna unit is mounted below the aft portion of the main rocket motor, as seen in Fig. 1.

Climatization. In the payload arrangement, the climatization system, discussed on page 23, requires little or no special consideration other than the

location of the radiator surfaces required for heat rejection from the water coolant. It appears that two radiating surfaces, approximately 4 by 7 ft each, are required (see Table 2). These may be located on the port and starboard side of the main rocket motor and are exposed following the booster separation.

Propulsion

The selection of a propulsion system for a satellite in the time period considered here is based on the following factors: performance reproducibility, system reliability, and powerplant availability. Contrary to experience with normal missiles, such items as propellant cost, handling characteristics, and production potential are reduced to minor importance, since only a small number of vehicles is needed. Because several contemporary propulsion systems may satisfy satellite performance requirements, the performance reproducibility and system reliability become more important than powerplant availability in making a choice of systems. Further, as noted under "Program Considerations" in Vol. I, the development schedule for a satellite is a direct function of powerplant availability; therefore, emphasis is placed upon selecting a system which is based, to a large extent, on present-day techniques.

A survey of available rocket motors shows that the gasoline and liquid-oxygen combination is to be preferred from the standpoint of combining adequate performance with most of the desirable characteristics mentioned. A chamber pressure of 600 psia is assumed for both stages, together with an oxidizer-to-fuel-weight ratio of 2.27[1]* and nozzle-exit-area ratios of 10 and 20 for the initial and final stages, respectively. The variation of specific impulse with altitude is noted in Fig. 2. The complete expansion value of sea-level impulse is 274 sec for 600-psia chamber pressure and optimum mixture ratio.

Booster Propulsion Considerations. The satellite has a sea-level booster-take-off thrust requirement of 284,660 lb. The total system consists of two large fixed units having 120,000 lb of thrust per motor and two gimbaled units having 22,330 lb of thrust per motor. All four units are regeneratively cooled with gasoline. While a single motor capable of fulfilling the take-off thrust requirements can be developed, such a unit is not considered here because a unit resulting from such a change in the state of the art will not be available, with sufficient reliability and reproducibility, within either the time period or costs considered. In addition, studies by several agencies, including RAND, have indicated a decrease in the thrust weight ratio for extremely large motors.

*For references, see p. 111.

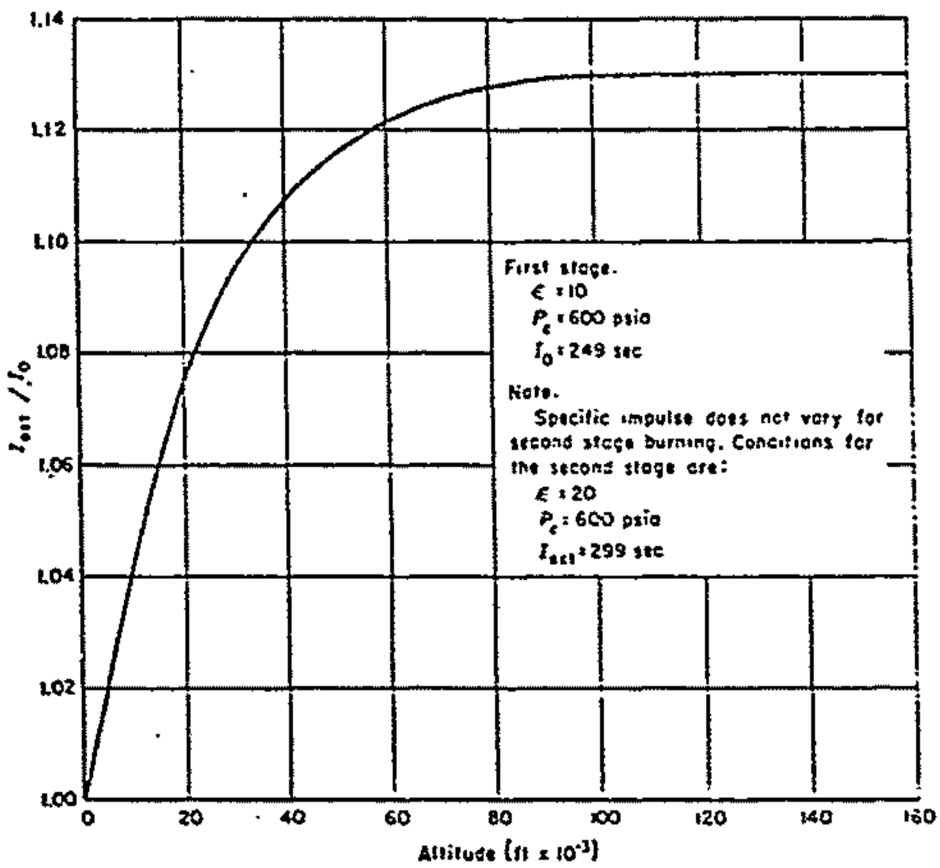

Fig. 2—Variation of specific impulse with altitude during first-stage burning

Propellant feeding is achieved through the use of a turbopump system, the turbine being driven by a gas generator using the main propellants as an energy source.

Tank pressurization is maintained at 40 psi gauge for structural integrity. This internal pressure, combined with the additional pressure resulting from the vehicle dynamics, results in a pump inlet head which is considerably larger than that required to prevent pump cavitation under normal operating conditions. Initial pressurization is obtained by means of inert gas and pressurization during flight and is maintained by means of propellant evaporation in a heat exchanger.

Satellite-stage Propulsion Considerations. In over-all characteristics, the main propulsion unit of the satellite stage is quite similar to the booster system. Because the thrust requirements are considerably less, a single motor unit having a 36,000-lb thrust is used. Although the final-stage propulsion system includes four vernier propulsion units, jet vanes are used for control during primary

burning. While the use of these jet vanes results in a slight increase in weight, they are felt to be desirable because the problems associated with restarting the vernier motors after the coasting period are avoided. (Attitude during coasting is maintained by the flywheels mentioned on page 132 of Vol. I.)

The burning time for the satellite stage is several minutes long; therefore new techniques will have to be developed to ensure adequate vane life. Some basic work has already been accomplished in this direction,[2] and additional effort should be devoted to include the feasibility of cooled jet vanes.

Although the vehicle is essentially horizontal during the satellite stage of burning, the axial load factor and the propellant level are combined in such a manner that positive flow to the pumps will be maintained during powered flight.

Following the coasting period, an additional increment of velocity is required to meet final orbital conditions. Because of the difficulty of restarting the main propulsion motor for this purpose, four vernier motors are used. Separate high-pressure propellant tanks are provided, and flow is maintained with a high-pressure gas operating against a separating diaphragm to ensure positive liquid flow. The propellant surface cannot be pressurized directly owing to intermixing of the gas and propellant during the gravitationless coasting period.

The ammonia-fluorine propellant combination was tested at North American Aviation, Inc. A 3000-lb-thrust unit was used, but pressure was obtained from tank pressurization and no pumps were employed in the system. The possibility that a self-contained rocket motor of ten times this thrust will be available for use on Feed Back is ever present. If this motor were used for the second stage only, the over-all vehicle weight would be reduced 45 per cent.

Structure and Weight

Probably the two most important design parameters for a rocket vehicle are the payload weight and the ν value—i.e., the ratio of initial usable propellant weight to initial gross weight—given by the flight trajectory. With these parameters determined, it is possible to optimize vehicle weight with respect to many of the basic structural parameters. The value of ν required for a gasoline–liquid-oxygen propellant system has previously been determined to be 0.801 (see "Flight Mechanics," page 68). Thus, eight-tenths of the vehicle gross weight consists of the propellant consumed by the rocket motors. The design ν of the vehicle is estimated to be 2.5 per cent greater than the value of 0.801

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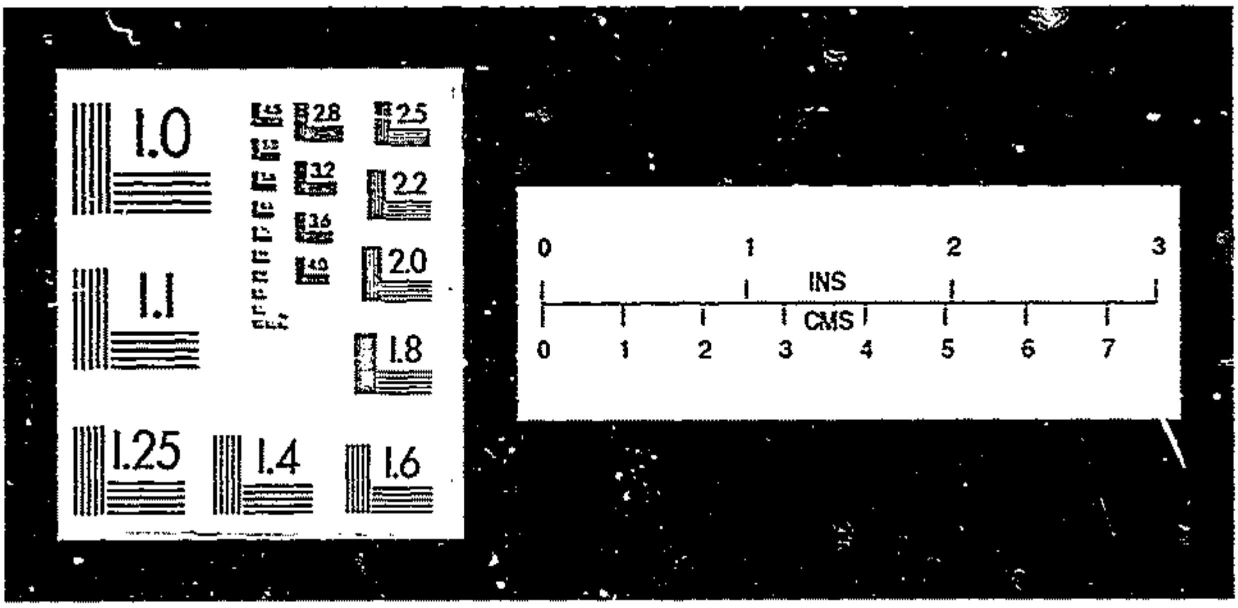

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required by the trajectory, because (1) it is not possible to use all the propellant aboard—owing to evaporation and to ullage and outage of the tanks, lines, and pumps; (2) some of the propellant is used as turbine fuel; and (3) oils and lubricants are required in the propulsion system.

Figure 3 depicts the ascent portion of flight as a function of range covered on the earth. The principal flight trajectory phases involved are the power-on booster phase, the satellite's second-stage burning, a coasting portion, and the final acceleration, with vernier control, to attain the desired orbital conditions.

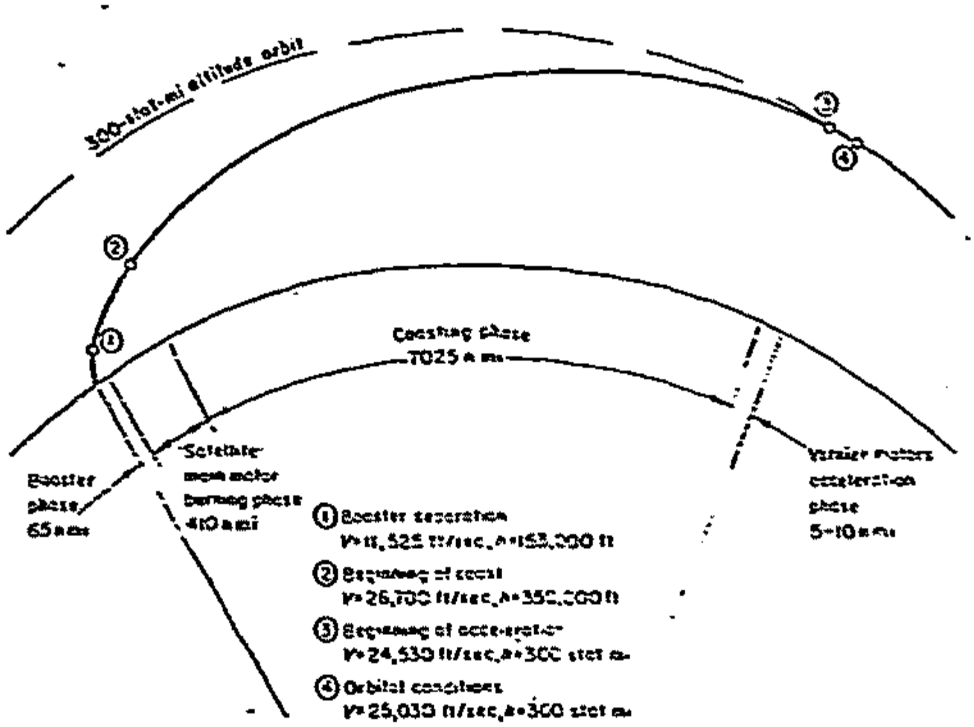


Fig. 3—Schematic of satellite's ascent trajectory for a 7500-mi ascent range

Further details of the primary power-on (precoasting) portion of the flight trajectory are shown in Fig. 4. The axial load factor—the ratio of vehicle acceleration to the standard acceleration of gravity at sea level—resulting from the above trajectory is shown as a function of flight time in Fig. 5. An initial load factor, n_i, of 1.60, defined as the ratio of powerplant thrust to initial gross weight, has been determined in previous RAND work as being nearly optimum for the two-stage configuration chosen. An increase in the axial load factor with time of burning results primarily from the linear decrease in vehicle mass

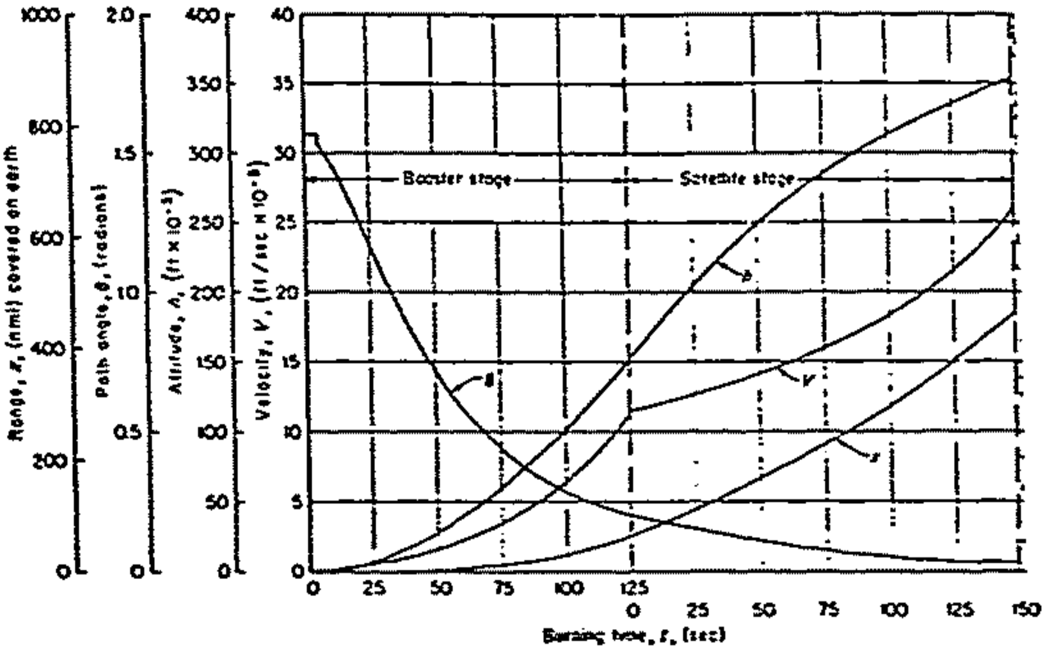


Fig. 4—Flight trajectory for the principal power-on (precoasting) portion of the satellite vehicle

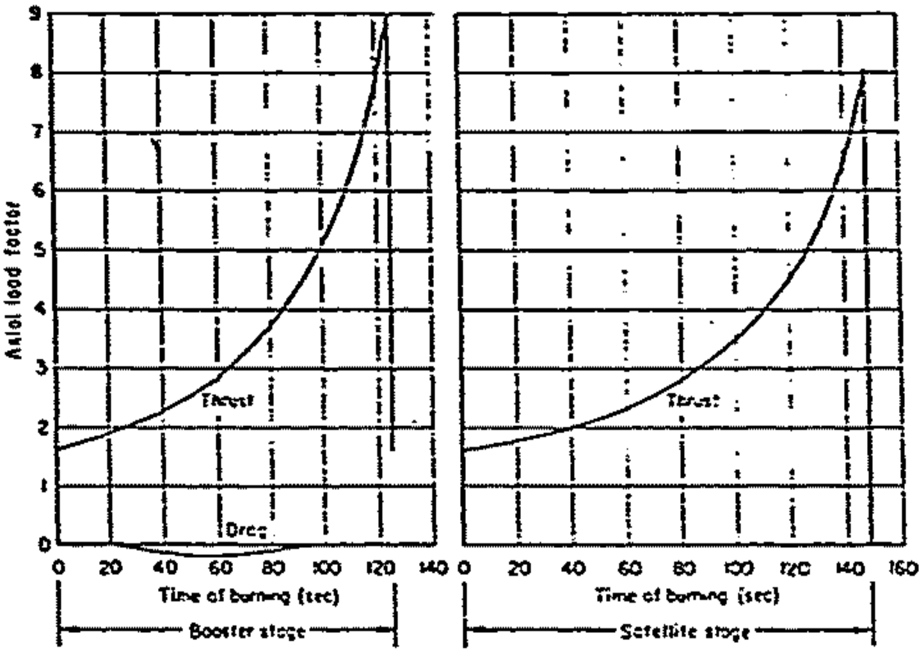


Fig. 5—Vehicle axial load factors resulting from axial acceleration

as the propellants are consumed. Also, because the axial load factor is a function of the specific impulse, I, of the propellant system, and because the specific impulse increases with altitude in the manner shown in Fig. 2, the final maximum load factor may be written as

$$\frac{n_i \times \dfrac{I_{\text{final}}}{I_{\text{initial}}}}{1 - \nu},$$

where the ratio of the final to initial specific impulse is approximately 1.13. This results in the development of approximately 9g's at rocket motor burnout. The axial compressive load experienced by the vehicle's primary structure owing to the above load factors is shown in Fig. 6 for the launching condition and at booster motor burnout. The vehicle flight path is a ballistic zero-lift turn into the desired 300-mi-altitude orbit. Launching is accomplished with the rocket in a vertical position, and the initiation of the gravity turn occurs a few seconds after rocket motor ignition. Because it is necessary to maintain the alignment of the vehicle's longitudinal axis tangent to the flight path, rocket motor control forces are exerted during boost to change the vehicle's attitude. In attaining the correct vehicle flight path by initial programming, aerodynamic loading occurs which introduces normal acceleration loads to the vehicle. In addition to the above-mentioned loads, an encounter with a sharp-edged gust will introduce sudden increases in lift, resulting in normal acceleration loads to the structure. Both of these normal acceleration forces are, of course, a function of (1) the dynamic pressures encountered by the vehicle, q; (2) the slope of the lift curve, $dC_L/d\alpha$; and (3) the ratio of the aerodynamic reference area to the instantaneous weight, A/W. The gust load factor is the product of the above parameters multiplied by the ratio of the gust velocity to the vehicle velocity. For the case of aerodynamic loads due to the control forces, the product of q, $dC_L/d\alpha$, and A/W must be multiplied by the vehicle angle of attack α. Figure 7 shows the variation of $dC_L/d\alpha$ with the Mach number for the satellite vehicle's configuration; the normal load factor due to a 60 ft/sec gust is shown in Fig. 8. This figure also shows the normal load factor experienced by the vehicle as a result of the aerodynamic forces which occur through the development of a 5-deg angle of attack.

Aerothermodynamic heating will cause rather high skin temperatures during

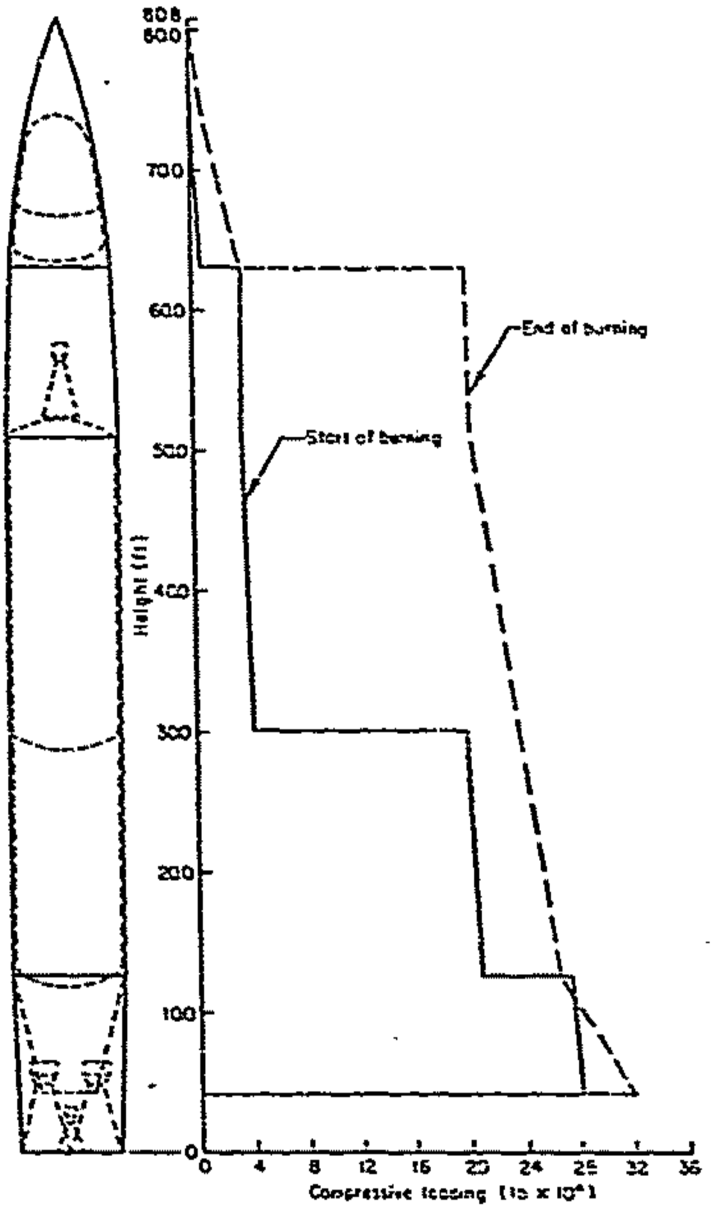


Fig. 6—Axial compressive loading diagram during boost

the powered flight, as shown in Fig. 14 on page 23, and the structure must be designed to resist the compressive loading at the reduced allowable stresses in the material associated with these temperatures. Inasmuch as it is not possible to predict with absolute certainty the boundary-layer transition during vehicle flight, the design skin temperatures for the stations where doubt exists are taken to be the temperatures reached when the boundary layer remains turbulent. (The temperatures will be lower under laminar conditions.) However,

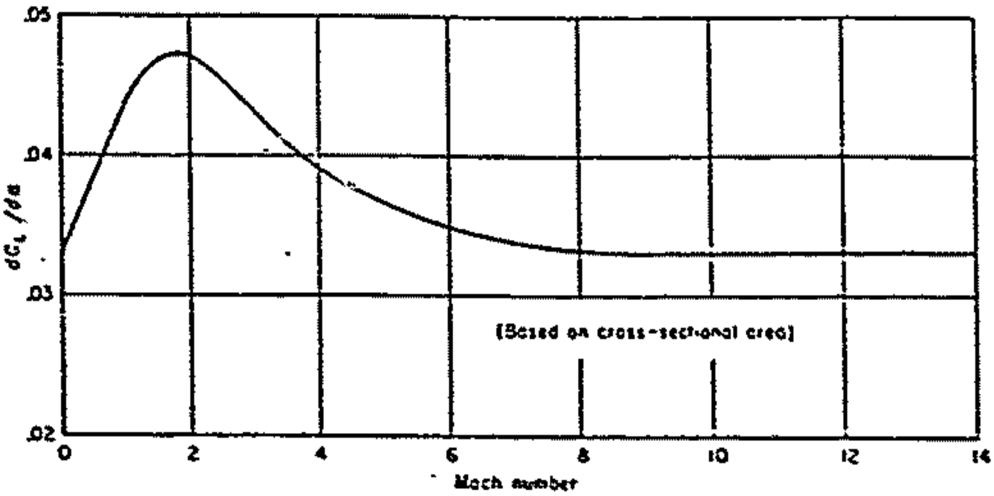


Fig. 7—$dC_L/d\alpha$ at angle of attack for zero lift vs Mach number for the satellite launching configuration

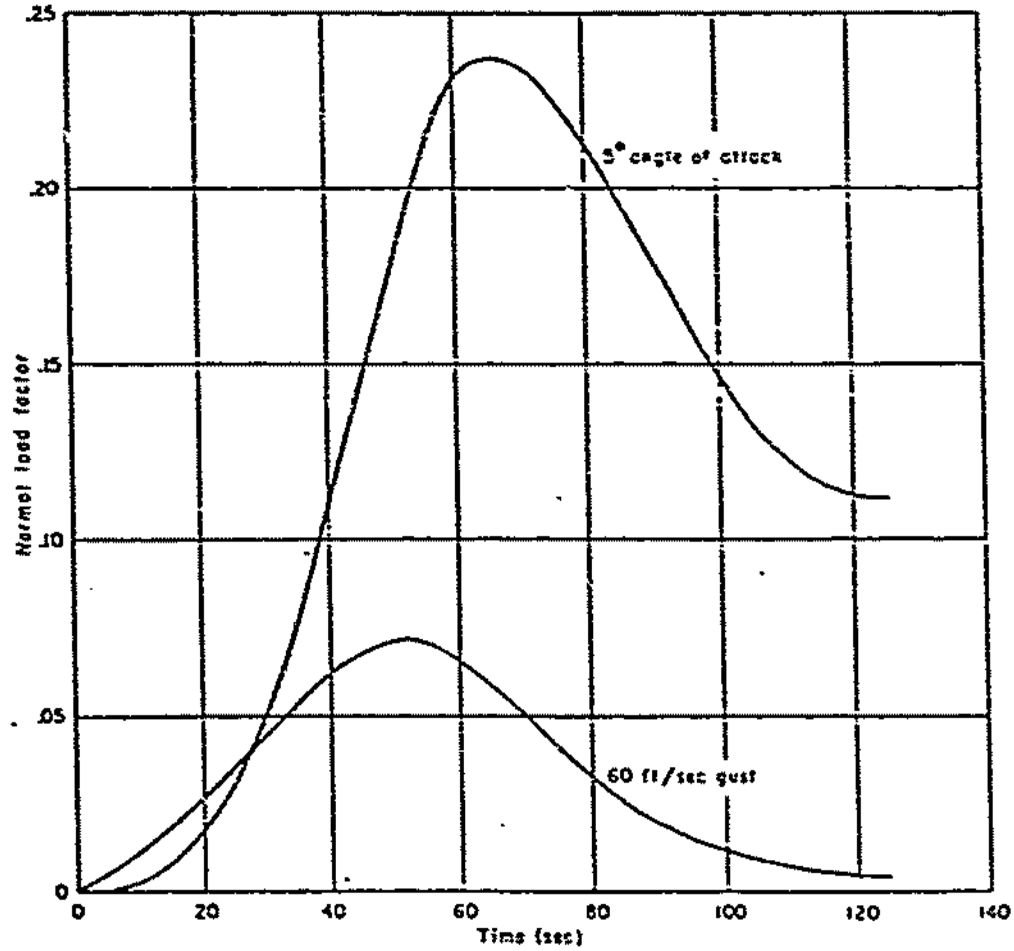


Fig. 8—Normal acceleration load factors resulting from gusts and control

owing to uncertainties in data, a possibility exists that temperatures higher than those we have used will be reached.

Primarily because of the high skin temperatures encountered in the satellite stage (see Fig. 14) during the ascent, and, secondarily, because of the heat dissipation requirements of the satellite's reactor radiator, load-carrying propellant tanks, housed in a nonstructural minimum-gauge steel fairing are used in the satellite stage. A similar arrangement is used in the booster stage. It can be shown that the use of integral booster tanks results in only a slight (5 per cent)[13] over-all vehicle weight penalty. However, the possibility that higher-than-predicted skin temperatures will be encountered makes it desirable to incorporate insulation, which can be obtained by using a nonstructural covering over the tank section. Obvious advantages of such an arrangement are that propellant evaporation is reduced and that accessibility to wiring, hydraulic lines, and the reactor radiator in the satellite is facilitated. Furthermore, the use of unheated nonintegral propellant tanks as primary load-carrying members permits the full use of room-temperature material-strength properties.

Maximum compressive loading for the interstage support structure is seen from Fig. 6 to occur just prior to stage separation. Skin temperatures for this portion of the vehicle are represented in Fig. 14 as the curve for the booster. At stage separation the temperature reaches its peak value of 1160°F.

Structural Design. As stated in Ref. 3, the over-all vehicle configuration is an ogive-cylinder-boattail combination, with an attachment between the two stages at the aft end of the second-stage propellant tanks. A structural covering over the second-stage powerplant compartment serves to transmit the axial acceleration loads during boost; it is integral with the booster and is carried away at the staging separation. Principal dimensions of the vehicle are shown in Fig. 10 on page 20.

A conventional semimonocoque construction is employed throughout both the satellite and the booster. All propellant tanks are fabricated of 75S-T aluminum sheet and are internally stiffened. The exterior covering of the vehicle is constructed of a ½-hard 18-8 stainless steel sheet stiffened by 75S-T aluminum frames and stringers. To reduce the heat transfer, a thin insulating material is required between the external skin and the stiffening elements prior to riveting the assembly. An exception to this procedure is found in the load-carrying fairing between stages, which is stiffened with steel frames and stringers. The satellite nose-cap temperature requires the use of a material such as Hastelloy C metal, and all motor mounts and related fittings are of SAE 4130 steel.

The manner of determining the vehicle's size and weight remains essentially as reported in Ref. 4. Pressurized internal tanks to transmit primary bending and compression loads, as previously noted, permit the full use of aluminum-alloy room-temperature material-strength properties. Considerable structural reinforcement is required at the tank ends in both the satellite and the booster stages because of bending moments imposed by the nonhemispherical tank ends and also because of the discontinuity of the load path at the junction of the tanks and the fairing between stages. The tanks are pressurized to 40-psi gauge, which provides ample margin over that required to resist the axial compression and bending loads experienced by the vehicle. To ensure structural integrity of the tank section, the design pressure is taken to be 15 per cent greater than the combined internal gas pressure and the acceleration head of the propellant. Design internal pressures of the vehicle's booster tanks, together with the pressure required structurally to resist the axial compressive loads, are shown in Fig. 9.

Weight allowance is made for nonoptimum sheet-thickness gauges, access doors, inspection openings, local reinforcements, joint efficiencies, and material physical-strength reduction resulting from welding. The amount of secondary structure necessary for the attachment of component items, such as brackets, gussets, fittings, supports, etc., is approximated, consideration being given to the proximity of the component items to the primary structure.

Table 3 presents a weight summary for the 1500-lb-payload satellite vehicle using gasoline–liquid-oxygen propellants. The gross weight of the vehicle at launching is seen to be 177,905 lb and the dry weight to be 15,360 lb. The satellite at stage separation has a gross weight of 22,520 lb and an orbiting weight of 4480 lb. Figure 10 is a sketch of the vehicle configuration and includes a table of the various weights by vehicle components.

To check these weights, component weights of the satellite vehicle are compared with those of existing or proposed rocket missiles. One way to do this is to examine rocket performance parameters. As discussed under "Propulsion," above, the rocket-motor propulsion package weight is in keeping with the present rocket-motor performance and trends and appears to be quite realistic.

The booster propulsion package consists of two fixed 120,000-lb-thrust rocket motors with two gimbaled 22,530-lb-thrust rocket motors. Figure 11, a plot of rocket motor thrust-to-weight ratio as a function of the thrust developed for existing liquid-rocket motor designs,[9] indicates the weight of the

above four rocket motors to be 1150 lb. The remaining weight of the total propulsion system, for the same liquid-rocket propulsion packages used in the curve in Fig. 11, is shown in Fig. 12 as a function of the following parameter: propellant flow rate × (time of burning)$^{\frac{1}{2}}$.

The weight of the booster propulsion system of the satellite (minus rocket motor) is approximately 3650 lb. Thus these curves, representing present-day trends, indicate that the total weight of the propulsion system is 4800 lb. Table 3 shows that the booster's total propulsion package (exclusive of gimbal mounts, attachments, and thrust mounts) weighs 4435 lb. The weight reduction represented here is that which is likely to result from future design im-

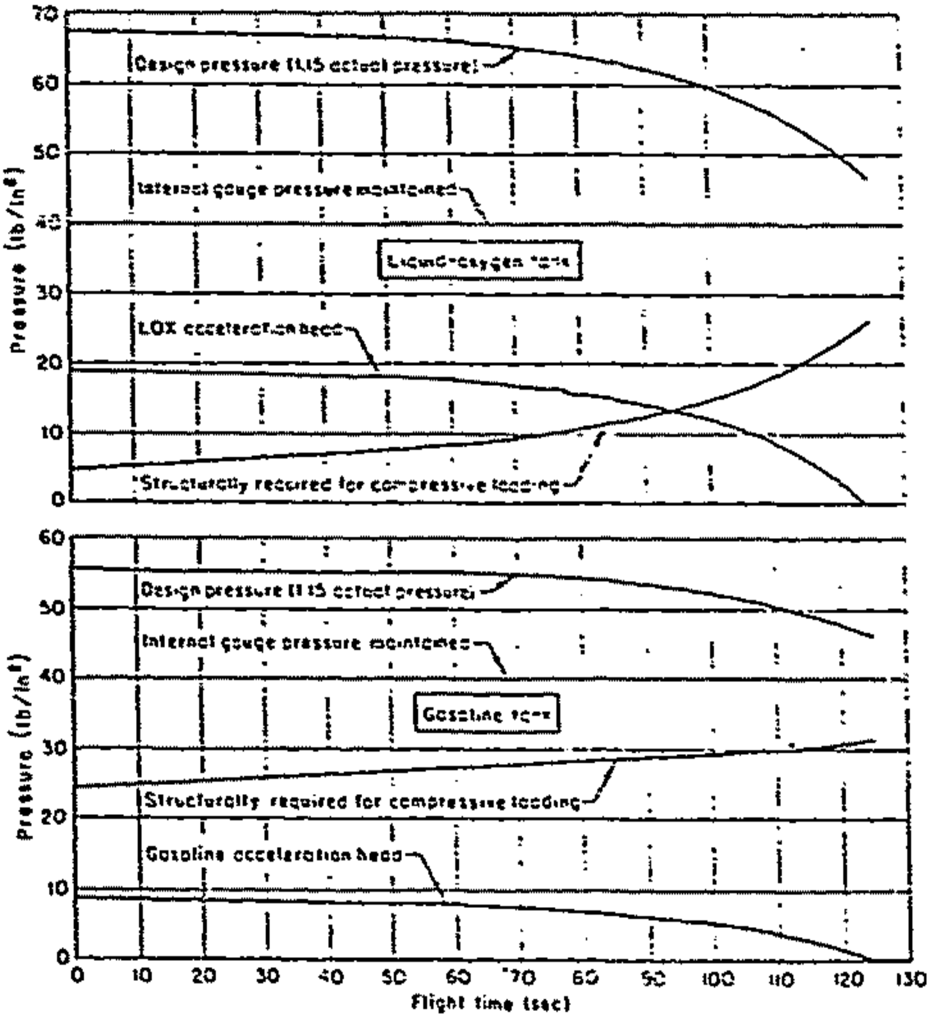


Fig. 9—Internal design pressures at base of booster propellant tanks

Table 3

WEIGHT BREAKDOWN OF THE SATELLITE VEHICLE
(In pounds)

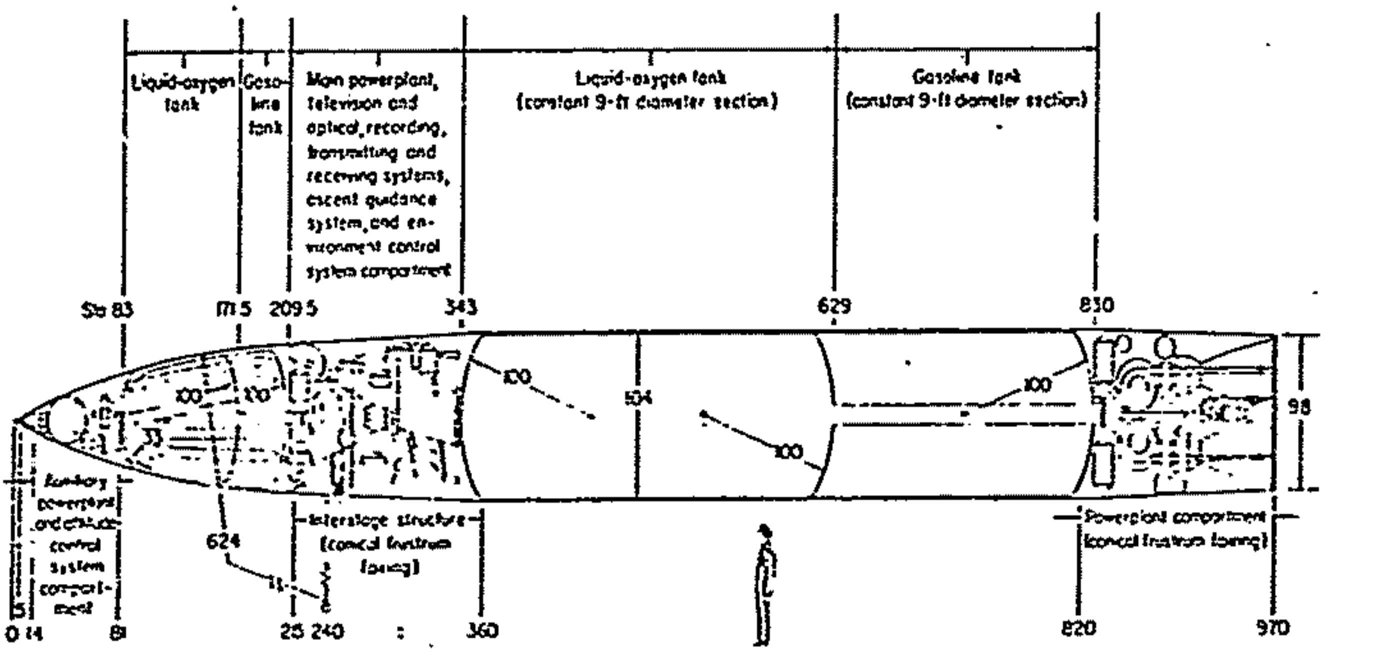
SATELLITE STAGE

Payload			1,475
Ascent guidance and attitude controls		280	
Television and optics		270	
Recording equipment		205	
Transmitting and receiving		140	
Auxiliary power source		500	
Environment control system		80	
Fixed equipment			225
Electrical and electromechanical system		125	
Hydraulic system		100	
Structure			835
Nose cap		10	
Sta. 14.3 to Sta. 90		115	
External skin	80		
Stiffening elements	35		
Propellant tanks		345	
Oxygen tank walls	70		
Oxygen tank ends	37		
Oxygen tank stiffening elements	65		
Gasoline tank walls	33		
Gasoline tank end	27		
Gasoline tank stiffening elements	53		
Tank skirts	60		
Outer shell		305	
Sta. 90 to Sta. 215 external skin	225		
Stiffening elements	80		
Secondary structure		30	
Insulation		30	
Propulsion system			1,495
Rocket motor powerplant		200	
Propulsion accessories		755	
Electrical and pneumatic system		85	
Jet vanes		90	
Vernier motors		40	
Vernier motors accessories and system		175	
Motor mounts and attachments		150	
Total propellant and oils			18,490
Nonusable propellant oils and lubricants		450	
Usable propellants		18,040	
Liquid oxygen	12,525		
Gasoline	5,515		
TOTAL WEIGHT			22,520

Table 3—continued

BOOSTER STAGE

Payload (satellite stage)			22,520
Structure			3,710
Booster-satellite fairing		545	
External skin	260		
Stiffening elements	210		
Stage-separation structure	75		
Propellant tanks		1,440	
Oxygen tank walls	425		
Oxygen tank ends	105		
Oxygen tank stiffening elements	210		
Gasoline tank walls	270		
Gasoline tank end	60		
Gasoline tank stiffening elements	130		
Forward tank skirt	35		
Forward tank skirt stiffening elements	15		
Aft tank skirt	60		
Aft tank skirt stiffening elements	30		
Internal piping, ducts, and baffles	100		
Tank section covering		1,195	
External skin	935		
Stiffening elements and attachments	260		
Powerplant compartment covering		375	
External skin	255		
Stiffening elements and attachments	120		
Secondary structure		80	
Insulation		75	
Propulsion system			5,095
Rocket motor powerplants		1,085	
Propulsion accessories		3,070	
Electrical and pneumatic system		280	
Gimbal mounts		60	
Motor-thrust mounts and attachments		600	
Fixed equipment			525
Electrical and electromechanical system		125	
Hydraulic system		250	
Stage-separation mechanism		150	
Total propellant and oils			146,055
Nonusable propellant, oils, and lubricants		3,555	
Usable propellants		142,500	
Liquid oxygen	98,925		
Gasoline	43,575		
TOTAL LAUNCHING WEIGHT			177,905
TOTAL DRY WEIGHT			13,360



Vehicle characteristics	Booster plus satellite stage		Satellite stage	
	Full	Empty	Full	Empty
Weight (lb)	177,500	35,400	22,320	4,480
Primary structure weight (lb)	2,485		620	
Covering and secondary structure weight (lb)	1,750		335	
C.G. location, station number	524	[illegible]	168	207
Mass moment of inertia, pitch and yaw (slug ft^2)	1,447,102	675,558	12,320	6,186
Mass moment of inertia, roll (slug ft^2)	49,405	7,605	3,393	495
Thrust developed (lb)	284,000*		36,030	
Propellant flow rate (lb/sec)	1,140		122	
Motor exit area to throat area ratio	10		[illegible]	
Combustion chamber pressure (psia)	600		600	
Propellant system	$O_2 + CH_{2.1}$		$O_2 + CH_{2.1}$	
Specific impulse (sec)	271.4†		299	

* Sea level static   † Integrated time average value

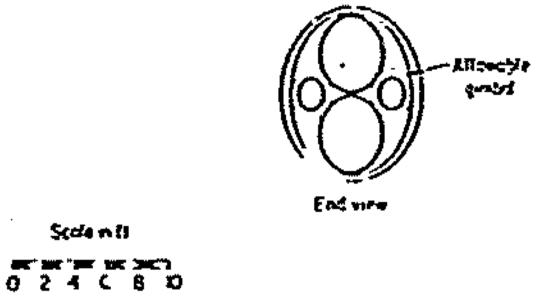


Fig. 10—Schematic of satellite vehicle

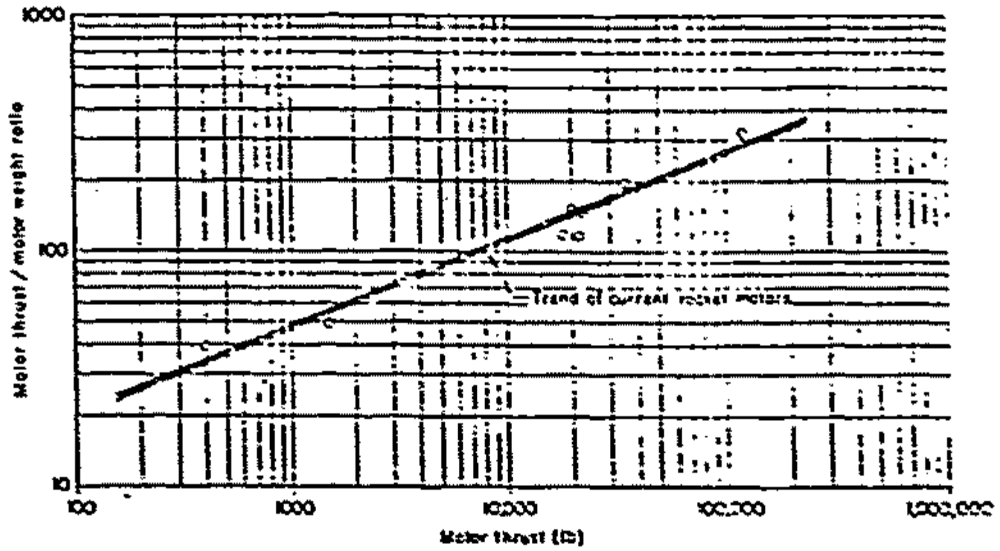


Fig. 11—Liquid-rocket motor thrust/weight ratio vs motor thrust

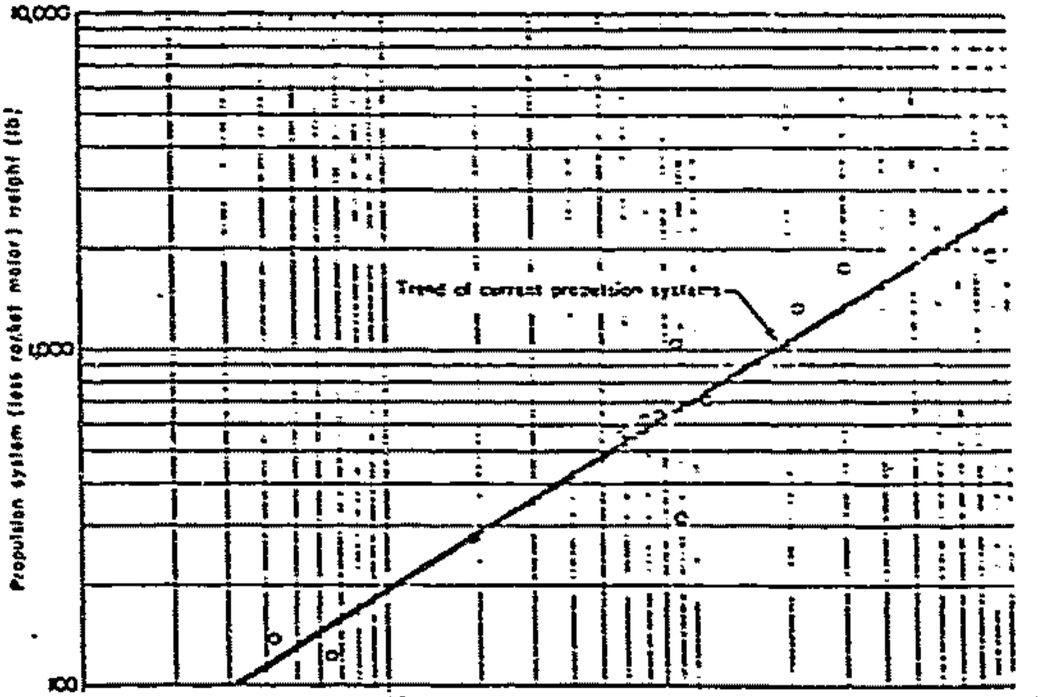


Fig. 12—Liquid-rocket propulsion system (less rocket motor) weight as a function of propellant flow rate times $(t_b)^{1/2}$

provements. Similarly, propulsion package weight items in the satellite stage will weigh 1100 lb according to the present-day trend shown in Figs. 11 and 12, but, from Table 3, these items will total 1040 lb for a future design.

A second performance parameter which may be examined is vehicle empty weight as related to rocket motor total impulse. Total impulse (defined as the maximum thrust developed times the effective time of burning) is an indication of the vehicle's velocity potential, and the empty weight is a measure of the efficiency of producing the total impulse. Empty weight here is defined as comprising the weights of all items, except propellant, contributing to packaging and production of vehicle thrust. This includes the tank, fairing, and motor-mount structure, propellant system, rocket powerplant, controls, trapped and evaporated propellants, and all oils and lubricants. Figure 13 presents a plot of the total impulse of several current and proposed rocket missiles as a function of the empty weight of a missile performing the boosting function.[3] The components of the satellite booster empty weight, plotted as a function of the booster total impulse, are shown to be in close agreement with the trend established by the curve. The second-stage satellite, when considered as a boosting stage, may also be seen to give good correlation with the trend curve.

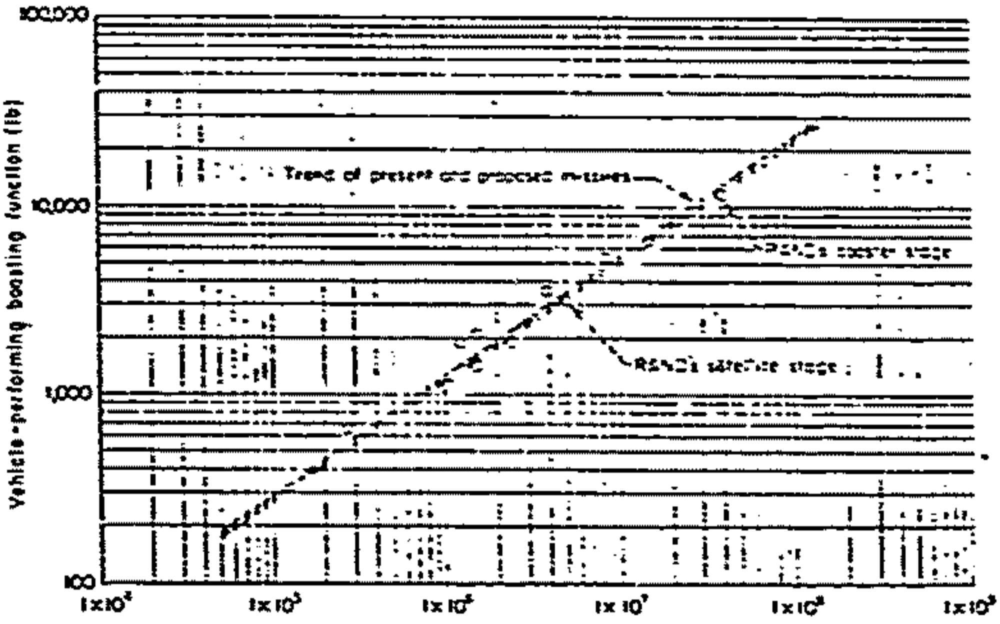


Fig. 13—Vehicle empty weight as a function of vehicle total impulse

Skin Temperatures during Ascent. An increase in the skin temperature occurs during ascent through the atmosphere because of aerodynamic heating, i.e., conversion of kinetic energy to thermal energy in the inner regions of the boundary layer. These high temperatures dictated the choice of the booster double-skin construction described under "Structural Design," above.

The skin temperature as a function of time, as computed by the methods given in Ref. 6, is shown in Figs. 14 and 15 for various stations on the satellite and booster. In the case of the satellite nose projecting ahead of the radiator, the skin thickness was adjusted (temperature-strength considerations) so that the stresses were reasonable. Note that the temperature of the auxiliary power-plant radiator remains essentially constant. This was accomplished by adjusting the LOX tank insulation thickness so that the heat absorbed by the LOX and lost by radiation was balanced by the aerodynamic heating and the heat given up by the radiator fluid. The outer skin of the booster stage will rise appreciably

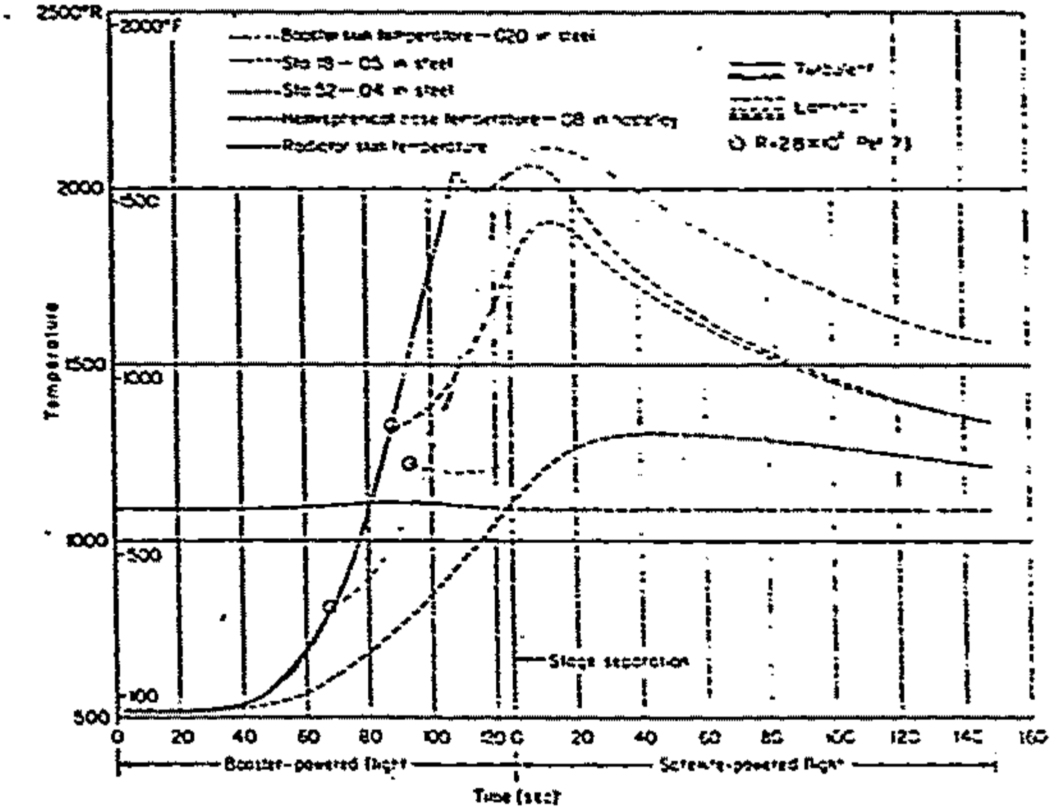


Fig. 14—Skin temperature of a satellite during powered flight—1500-lb payload, 300-mi-altitude orbit

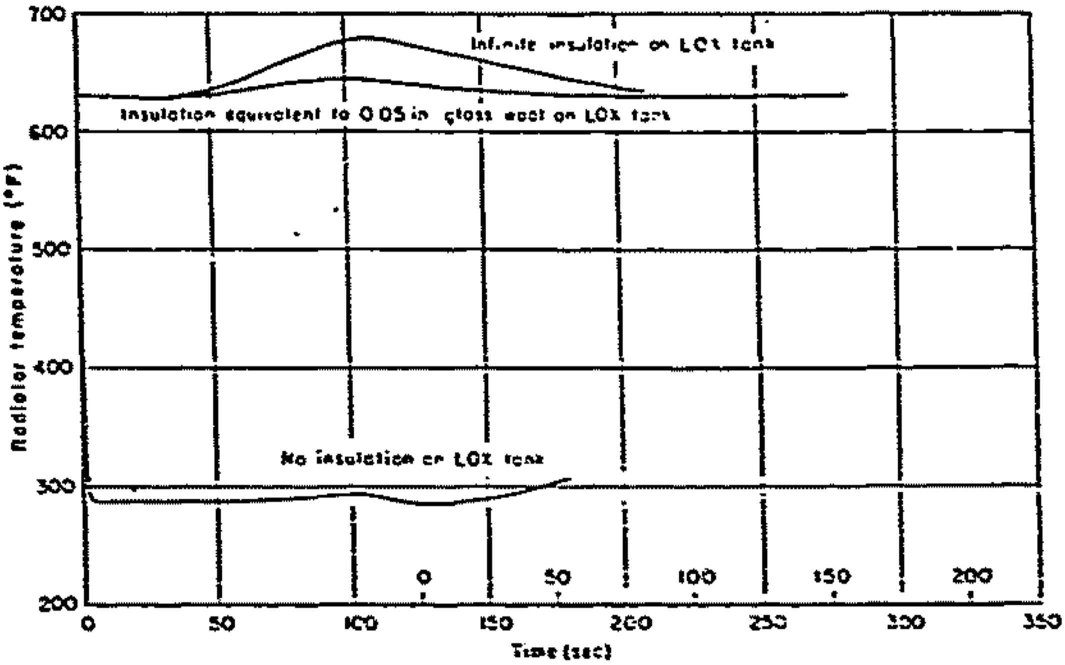


Fig. 15—Variation of average radiator temperature during powered flight—0.020-in. steel skin

in temperature, but this skin supports only the local aerodynamic loads, so that the temperatures shown can be tolerated.

Because of the uncertainty of the criteria for boundary-layer transition,[16] some of the skin temperatures are shown both for an essentially all-turbulent condition and for transition to a laminar boundary layer as indicated by the transition criteria adopted for previous RAND studies.[17] At present it is not possible to predict with confidence the conditions for boundary-layer transition, although the theories of laminar boundary-layer stability indicate that there are certain ranges of Mach number and surface temperature where the boundary layer is stable. According to this criterion, the boundary layer over the satellite and booster may be turbulent (i.e., unstable) during the entire time when convective aerodynamic heating is appreciable, except over the radiator section, which (because of its lower temperature due to internal heat transfer) is predicted to have a laminar (stable) boundary layer above a 50,000-ft altitude.

Cooling of Electrical Equipment. For the proper operation of the electrical equipment in the satellite, temperature control must be provided during both the ascent and the orbiting phases. The control selected (shown schematically in Fig. 1 on page 5) consists of a water-circulating system to carry the heat from the points of heat generation to radiators on the outer periphery of the vehicle.

The heat to be dissipated is about equal to the electrical output of the powerplant, i.e., 2 kw. Two radiators, each 4 ft by 7 ft, are sufficient to dissipate this heat (see Table 2).

No attempt has been made to make detailed designs of the cooling of each individual component. However, a suitable conduction or radiation path must be provided for the transfer of heat from the heat sources in the component to the cooling coil or cooling jacket provided for that component. A variety of such designs has been given by Robinson.[13]

Radiators for the cooling system are located on either side of the motor, on the periphery of the vehicle. They lie under the forward skin of the booster stage and will be exposed at the time of separation. The cooling system has sufficient thermal capacity to absorb the heat generated by the electrical equipment before the time of stage separation.

Skin-temperature Variation in Orbit. At the orbiting altitude of 300 mi, the atmosphere is so tenuous that the skin temperature is determined solely by radiative heat transfer. Atoms and ions of nitrogen, oxygen, hydrogen, and helium are the particles to be found in the atmosphere at this height. Even if chemical association of the atoms during adsorption occurs, the heat generated is small compared with the heat transferred by radiation. The only probable effect of the "air" at the 300-mi altitude is a reduction of the metallic oxide on the surface of the satellite skin caused by the atomic and ionic hydrogen and nitrogen. Since the thermal emissivity and absorptivity of a metal is very sensitive to surface condition, this effect will be important to the surface temperature. No attempt will be made here to predict what the net result of the rarefied atmosphere on the surface will be; however, orbit temperatures have been estimated for both oxidized and unoxidized metal skin.

If the surface radiation characteristics (absorptivity and emissivity) are known as a function of the radiation wavelength, the temperature variation of a given surface element on the vehicle may be readily computed by methods presented elsewhere.[14]

Figure 16 shows, for one revolution around the earth, the variation in surface temperatures on the top, bottom, and side of a vehicle having a 0.020-in. stainless steel skin in both a "clean" and an oxidized condition. For both surface conditions, an appreciable temperature variation occurs as the satellite passes over "day" and "night" on the earth. The largest variation occurs for the "space" side of the vehicle, since it is exposed to the greatest variation in conditions. In the case of an oxidized surface, the average (positionwise) skin temperature cycles around a temperature of about 66°F. Internal components

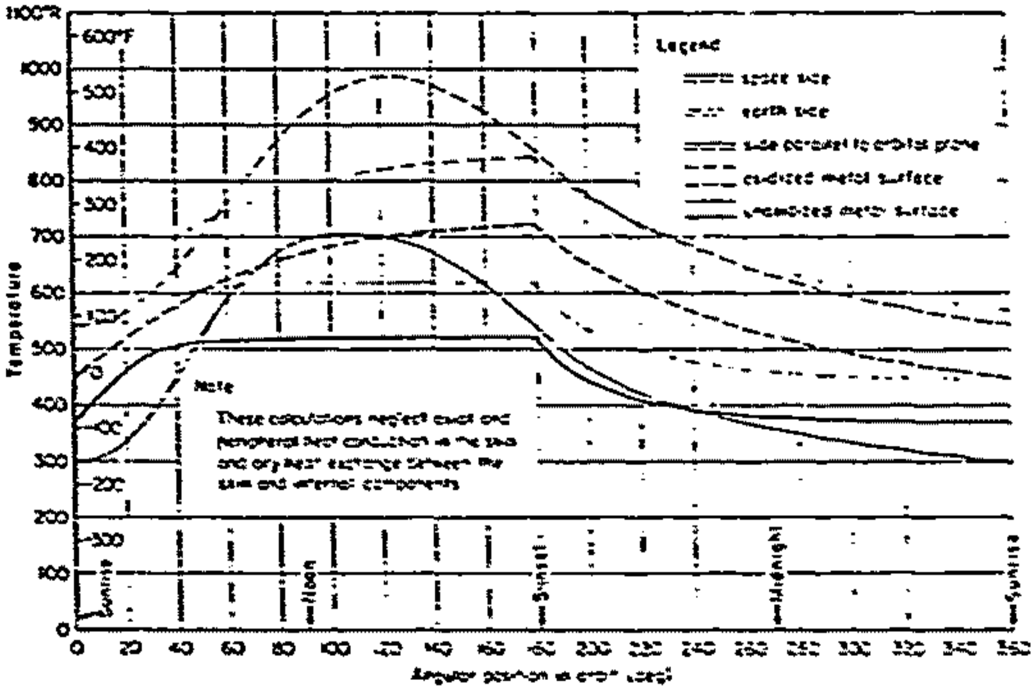


Fig. 16—Skin temperature variation of a satellite on 300-mi-altitude orbit—0.020-in. stainless steel skin

will cycle about a slightly higher temperature (assuming their own heat generation is removed by the cooling systems), but with a reduced amplitude. Thus a tolerable temperature level is provided—if the skin can be kept in the oxidized condition. The mean temperature of the unoxidized skin, for instance, is about 250°F. Such a temperature will prevent the operation of most of the electronic equipment and will require a much larger powerplant radiator. However, this condition can be greatly alleviated by deliberate sandblasting, coloring, the electrodepositing of nickel black, etc. The importance of further investigation of the effects of the dissociated and ionized atmosphere on a metal surface is indicated.

Diffusion of Gases through Metal Walls. Because of the extremely low ambient pressure during orbiting conditions, diffusion of gases through the walls of any pressurized component must be considered. Table 4 indicates the diffusion rate in several gas-solid systems for a 1-atm-pressure difference across a 0.020-in. wall. It is apparent that at normal temperatures the diffusion rate is extremely small, and therefore no difficulty should be expected (from diffusion) in order to maintain pressurization of, say, a piece of electronic equipment. However, at higher temperatures, particularly where hydrogen is used (e.g., in an auxiliary powerplant), the diffusion rate is quite appreciable. Pro-

vision must be made, therefore, to reduce the diffusion of hydrogen through the walls of its containing tubes. One possibility appears to be the use of a glass or ceramic coating. As Table 4 indicates, the diffusion rate through glass is about three orders of magnitude lower than that through metal in the higher temperature range. A further possibility would be the use of a hydrogen storage tank to replenish the hydrogen lost by diffusion.

Table 4

DIFFUSION OF GASES THROUGH METAL WALLS

Diffusion Rate: lb/yr (ft²)

System	80°F	440°F	800°F	1540°F
H_2-Al	3.07×10^{-13}	4.44×10^{-8}	3.22×10^{-3}	2.68
H_2-Mo	5.57×10^{-12}	4.35×10^{-4}	1.46×10^{-3}	0.114
H_2-Ni	5.71×10^{-7}	4.64×10^{-3}	0.222	3.94
H_2-Fe	5.15×10^{-4}	3.32×10^{-2}	0.519	4.13
N_2-Fe	8.67×10^{-4}	7.71×10^{-7}	7.22×10^{-4}	0.125
CO-Fe	1.58×10^{-10}	4.15×10^{-3}	8.88×10^{-2}	0.490
H_2-SiO_2	1.20×10^{-7}	3.88×10^{-5}	4.66×10^{-4}	2.99×10^{-3}

Another factor which should not be overlooked is the embrittlement of steel by hydrogen at high temperatures.

B-1
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page
35

connection with the satellite's auxiliary powerplant. The Rankine, while not necessarily the optimum, does appear to be one feasible solution, based on the present state of the art.

Any thermodynamic cycle used for power production involves compression of the working fluid, heating it at the higher pressure, and subsequent expansion through some sort of engine (a turbine, for example). For use in connection with the Feed Back device, much depends on the way in which the working fluid is compressed. Processes involving boiling, or those using large volumes with regenerators so that compression can take place at a leisurely pace,* have nearly 100 per cent efficient compression. The former method is intrinsic with the vapor (or Rankine) cycle, whereas the latter type of compression is a characteristic of "air" engines (the Stirling cycle, for example).

A simple gas cycle is the Brayton—that used by ordinary gas turbocompressors. However, in the case of the Brayton cycle, a large percentage of the work produced by the turbine goes toward powering the compressor. Therefore a small change in the efficiency of either component would result in no power output, because the latter would be the net difference between two large numbers. If adequate component efficiencies could be achieved, a hydrogen gas turbine would be attractive.

The Rankine cycle, on the other hand, involves but a small amount of pump work, and is much less dependent on high component efficiencies. As a result, it is the system considered here. In the main, it consists of mercury as the working fluid, a water-moderated reactor, a single-stage impulse type of turbine, a radiant condenser, and a mercury feed pump.

The choice of mercury as the working fluid was based on its favorable temperature-vapor-pressure characteristics. For example, at 500°F, its vapor pressure is 2 psia, and at 900°F the vapor pressure is 100 psia. Disadvantages associated with the use of mercury are its relatively low specific heat (0.0248 BTU/lb°F for superheated mercury vapor as compared with approximately 0.50 BTU/lb°F for superheated steam) and the high specific weight of liquid mercury (850 lb/ft^3).

Mercury conditions at the boiler outlet are 40 psia and 984°F (200°F superheat). Turbine exhaust conditions were determined primarily by the required temperature differential for heat transfer, turbine sealing requirements, and

*An infinite length of time to effect a thermodynamic change will yield a completely reversible, or 100 per cent efficient, process.

vapor quality necessary in the turbine blades. Turbine exhaust conditions of 10 psia and 657°F were regulated by the available energy across the turbine, the condenser radiation, and the pump suction-head requirements.

Neither the turbine nor the feed pump appears to offer any unusual problems. The problem of vapor quality is governed primarily by the mechanical design of the turbine.*

It is expected that rather unique but resolvable problems in the use of a two-phase substance under conditions of no gravity will be experienced, particularly in the case of the turbine itself. The collection of blobs of liquid or mixtures of liquid and gas in certain regions are examples.

One of the most important factors tending to regulate the power output of the auxiliary powerplant system is the availability of an adequate heat-rejection mechanism. Since the satellite vehicle operates essentially in a vacuum, the only feasible method available for long-time, steady-state heat dissipation for a closed cycle is external radiation. Thus, for any given vehicle configuration and auxiliary powerplant cycle, the maximum output power level of the reactor can be established in terms of the radiator heat-balance parameters.

Total heat to be dissipated from the radiator is the sum of the heat addition from external sources, i.e., irradiation from the earth and sun, and the internal heat due to the working medium. Reference 12 conservatively approximates the external heat input as a function of orbital position and surface emissivities. Because the purpose of this study is to determine feasibility, rather than to investigate the functional aspects of a detail design, the highest value for external heat input is used throughout. Combining the necessary heat rejection rates for the condensation of mercury (125 BTU/lb) with the flow rate (0.57 lb/sec), and allowing for a 10°F drop from the vapor (657°F) to the outer skin, the required radiating area is equal to about 135 ft^2 (see Table 2), or one-half the available lateral area of the satellite vehicle, for a 2-kw power output.

In view of the lack of available data on the condensation of mercury in a gravity-free medium, the resulting condensation heat transfer can only be approximated at this time. Based on preliminary calculations with conservative flow assumptions, it appears that condensation can be readily accomplished within the area limitations imposed.

*In accordance with the Becker and Doring theory,[11] the loss in turbine nozzle efficiency through condensation shock can be neglected.

B-1

Scanning

The scanning problem arises for an obvious reason: The limited size and resolving power of the Image Orthicon result in each picture's being able to contain only a finite number of bits of information. Elsewhere in this report it is shown that in order to keep the time between successive views of a particular ground area to a reasonable value, the television optical system must cover a strip extending for 200 mi on each side of the flight line. If this area were to be covered by a single picture, about 1 in. on a side, the scale would then be 1:25,000,000; if the spot size of the scanning beam in the camera tube could be kept down to 0.001 in., the image projected on the ground by the optical system would be 2100 ft in diameter. Anything much smaller than a mile in its principal dimension would be difficult to detect.

At the scale of 1:500,000, one picture is about 8 mi on a side. A strip 400 mi wide will require fifty pictures to cover it. This number of pictures must be transmitted in the time it takes the satellite to move forward 8 mi (1.68 sec), requiring a frame rate of about thirty per second, which is present commercial practice. At this scale, a spot 0.001 in. in diameter will cover a circle on the ground approximately 40 ft in diameter. Two television lines are equivalent to one optical line of resolution, and an object, to have a high probability of detection, must be covered by about two optical resolution lines.[23] This gives, in the present case, a limiting object size of somewhere between 150 ft and 200 ft, approximately the size of bombing aircraft—hence the gain realized by the complication of the addition of a scanning system.

Because the Image Orthicon is an integrating device, it requires a finite exposure time during which the image must remain fixed on the photocathode. (If it were not for this, the scanning problem would be reduced to the simple one of two cameras viewing the ground by reflection in continuously rotating mirrors. Two cameras would be required to eliminate the "dead time," i.e., the time during which the mirror into which each camera is looking would be re-

turned to its initial position to start a new sweep. But scanning in the direction of the line of flight is affected by the vehicle's motion. From the standpoint of reliability and long life, intermittent mechanisms which have been proposed for the projection of motion pictures from continuously moving film[24] are applicable. A few of those suggested in Ref. 24 may be difficult to fabricate, but they can be used successfully here, because many of the restrictions imposed by their application to theater projectors do not occur (e.g., the f number and back focal length, in particular, present no problems).

The most promising arrangement that has been investigated is the one proposed by RCA in their study of the problem. It consists of a number of mirror pairs mounted on the periphery of a continuously rotating wheel; each mirror pair deflects a ray through a fixed angle in a plane perpendicular to their line of intersection, independently of any rotation of the mirror pair about any line parallel to their line of intersection. In Fig. 29, M_1 and M_2 are two plane mirrors perpendicular to the plane of the paper, and I is their line of intersection. If the two mirrors rotate slightly, as a unit, the change of deviation of the ray produced by reflection in the first mirror is of the same magnitude and opposite sense as that produced by the second mirror, leaving the deviation produced by the pair of mirrors unchanged. The deviation of the pair depends, therefore, only on the angle between them. RCA's device is shown in Figs. 30 and 31. For a scale of 1:500,000, an altitude of 300 mi, and a strip 200-mi wide on each side of the line of flight, the rotating drum will have eighteen pairs of mirrors, each pair equally spaced around the periphery.

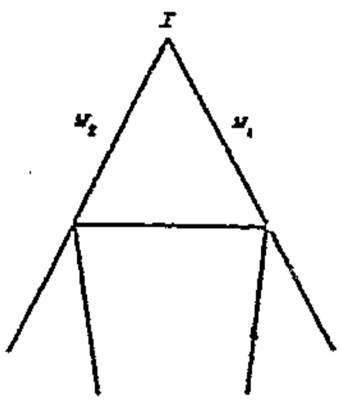


Fig. 29—Schematic of operation of mirror pair

This drum, whose axis is parallel to the direction of motion of the vehicle, rotates at a speed of 38.5 revolutions per minute in a clockwise direction as shown. Since there are eighteen equally spaced pairs of mirrors on the drum, the mirror pairs are spaced 20 deg apart. Both television cameras are spaced 90 deg apart as in Fig. 31. This means that at the instant that camera "A" is viewing a ground scene through a mirror pair, camera "B" is viewing the transition point between two successive mirror pairs. The sequence of ground scenes scanned is shown in Fig. 32.

Lateral image immobilization is achieved during the entire time that a pair, or part of a pair, of mirrors is in line with the optical axis of the camera. Except

Note-
Three of the 18 mirror pairs are shown schematically, each having a unique included angle (a_1, a_2, a_3, respectively) The optical system (lens A) of one television camera is viewing the portion of the ground scene presented by the a_1 angle, while the second camera (lens B) is simultaneously seeing the ground through the included angle a_2.

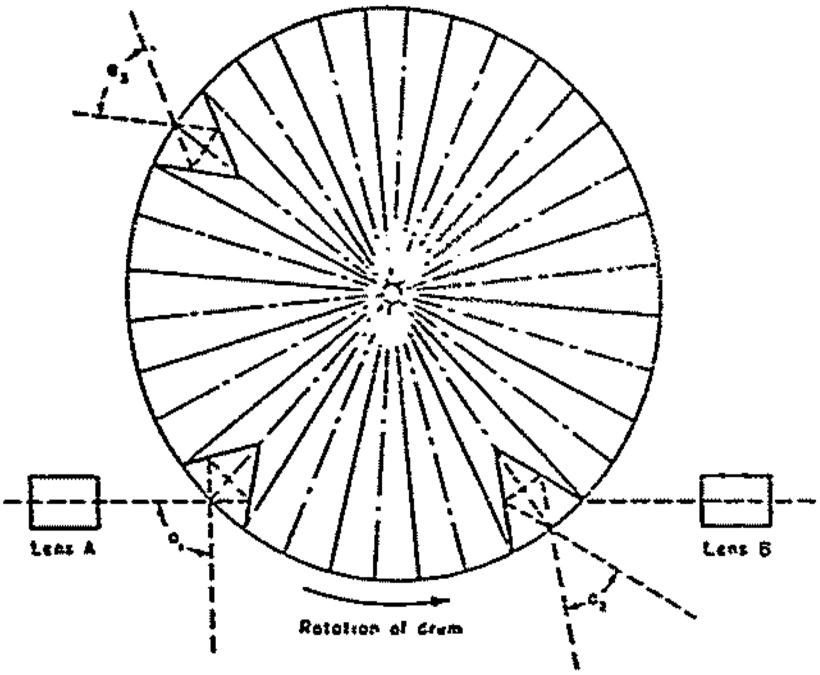


Fig. 30—Optical segmentation drum with four representative mirror pairs

for a short interval during which the image is recorded, a composite picture of two successive segments is seen because of vignetting effects. The situation perhaps can be explained better by describing the direction of view of one of the cameras. As the drum rotates, a scene, completely stationary except for the image motion caused by the forward motion of the vehicle (the lateral scanning introduces no image motion), can be observed for a period of time depending on the mirror size. The next scene will then be picked up and will start to blend with the first scene, both remaining completely immobilized. At some point, only the second scene will be observed; then the entire cycle will be repeated for adjacent fields of view.

Such a sequence is demonstrated by Fig. 33. The ordinate in this diagram is comparable to the intensity of illumination on the photocathode due to the fields of view indicated by the numbers above each peak, which can be made to correspond to the numbered fields shown in Fig. 31. RCA's proposal to avoid the resulting confusion of images is to "pulse" the image section of the Image

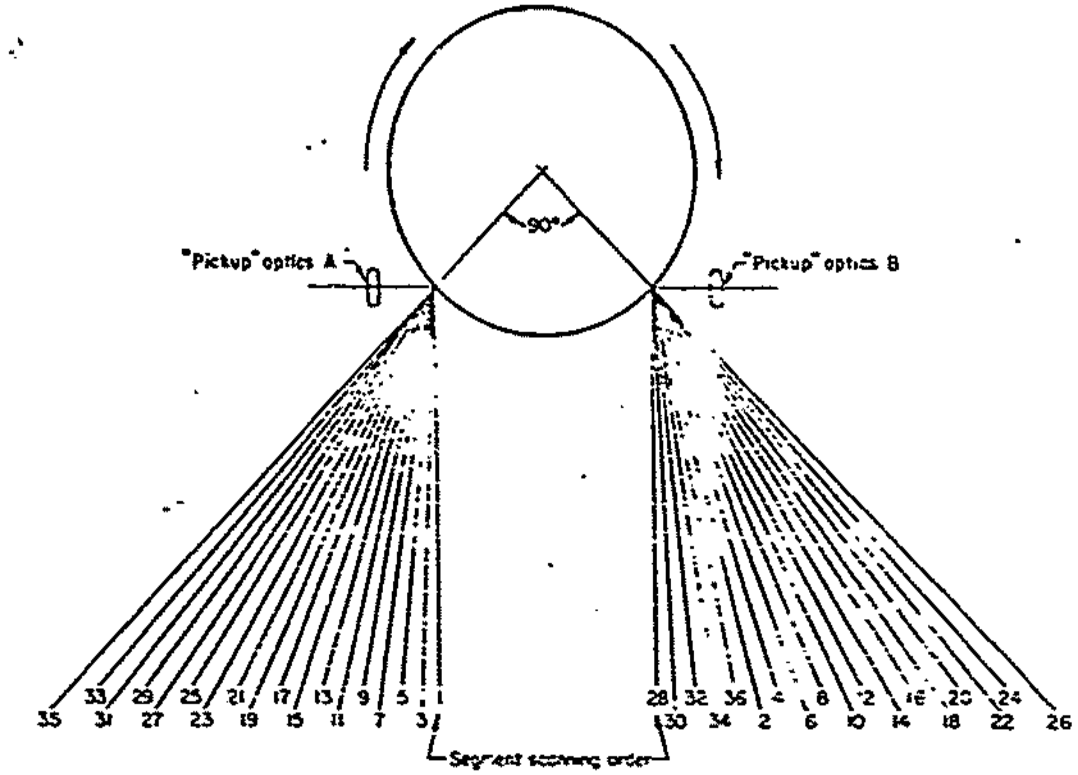


Fig. 31—Optical segmentation device as seen looking along the line of flight

Fig. 32—Scanning sequence

Orthicon. That is, the accelerating potential will be applied to the photoelectrons liberated by the optical image on the photocathode only during that part of the exposure on, say, field 3 when the light from fields 1 and 5 is less than some minimum value, say 5 per cent of full aperture. It should be noted that because the output of both cameras is to be transmitted over the same carrier wave and received on the same device, it is important that they be accurately interdigitated timewise. The curves in Fig. 33 represent the case where the dimensions of the mirror pairs are such that the full-aperture condition obtains only instantaneously. As the size of the mirrors is increased, the full-aperture exposure time increases and the exposure curves, shown in Fig. 34, develop

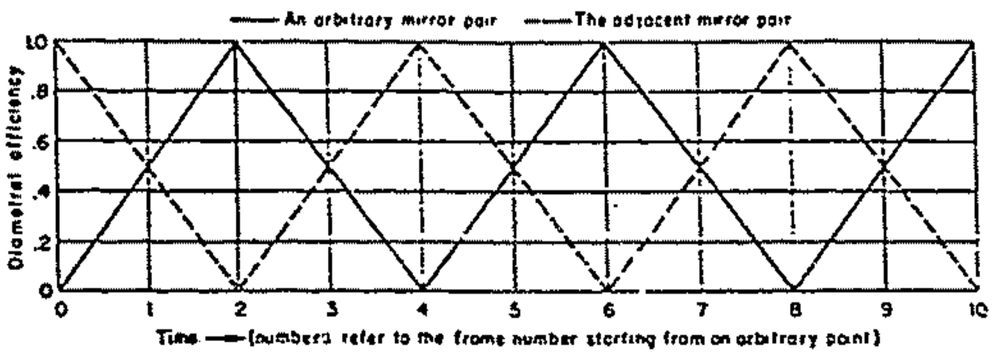


Fig. 33—Drum performance utilizing one set of optics and instantaneous exposure

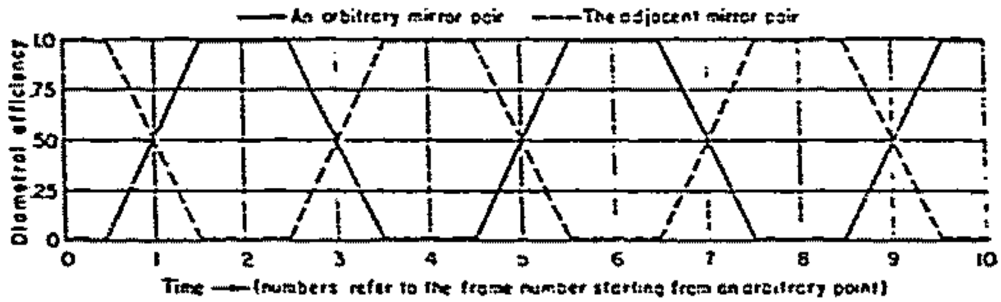


Fig. 34—Drum performance utilizing one set of optics and an exposure time equal to frame time

flat tops and bottoms. From the point of view of the most efficient time use, the optimum is reached when the exposure time is one-fourth of the frame frequency for both cameras; the exposure curve has the form shown in Fig. 34 for the camera on one side of the mirror drum.

Starting with field 2 (Fig. 34), full aperture is reached at the abscissa value of 1.5 and is maintained until 2.5. During this time the accelerating potential is applied and a charge image is built up on the target plate in the Image Orthicon. From 2.5 to 3.5 the image-stage voltage is shut off and the scanning beam discharges the image on the target plate. In this same interval (2.5 to 3.5) the camera on the other side of the mirror drum is being exposed. At 3.5 a new exposure is started in the first camera at the same time that the picture in the second camera is being scanned and transmitted. In this way there is always one and only one picture being exposed. There is no "dead time" for the transmitter.

The price that must be paid for this more efficient use of transmitter and exposure time is, of course, weight and bulk—the scanning drum must be larger. Variation of the drum radius with the percentage of the exposure time occurring at full aperture is shown in Fig. 35.

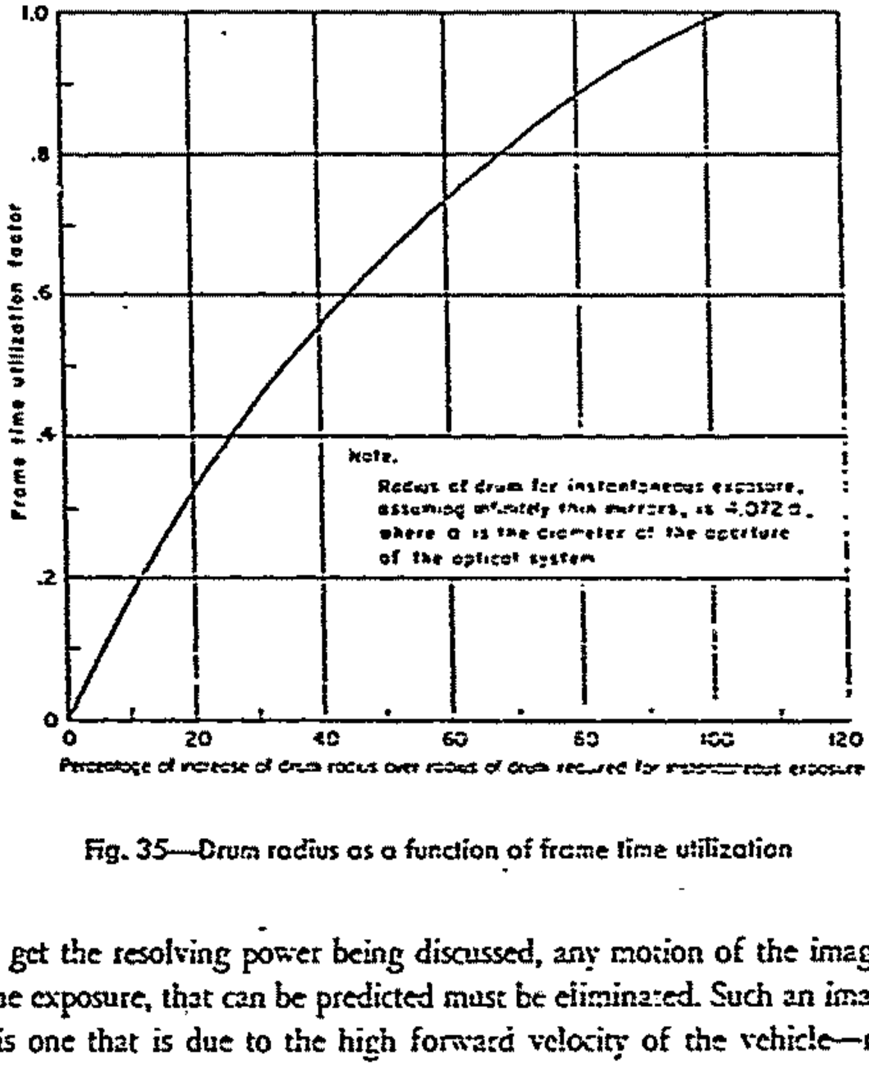


Fig. 35—Drum radius as a function of frame time utilization

To get the resolving power being discussed, any motion of the image, during the exposure, that can be predicted must be eliminated. Such an image motion is one that is due to the high forward velocity of the vehicle—roughly 25,000 ft/sec. At an exposure time of 0.001 sec, the image of the photocathode projected on the ground by the camera lens will move 25 ft, a barely tolerable amount. If the scanning mirror dimensions are chosen so that one-quarter of the frame time is available for exposure, the exposure time at 25 frames/sec will be 0.01 sec and the image motion during this time will be 250 ft, requiring some sort of image-motion compensation.

Here again the type of mechanism devised for the projection of motion pictures from continuously moving film can be used, but there is such a small motion, in terms of percentage of frame height (at most 1000 ft out of 8 mi),

to be corrected that, in RCA's opinion, it can be handled (electrically) by scanning in the image stage of the camera tube. All that is required is the addition of a coil above the image stage and one below the image stage, the plane of the coils being parallel to the axis of the tube. These coils can be energized by a "saw toothed" oscillator whose frequency is equal to the frame frequency. As the optical image moves on the photocathode, the photoelectrons liberated by a (moving) point in the image can be brought to a focus at a fixed spot on the target plate where the charge accumulates. RCA workers say that a motion of 5 per cent of the frame height is easily corrected in this way, whereas 10 per cent is possible to correct, but very difficult. Since the motion in this instance is around 2 per cent, it should not be difficult to correct.

The problem of obtaining reconnaissance data is essentially that of typifying various ground-target scenes with patterns of bits varying in intensity. The number of bits in a given period of time determines the bandwidth, or the information rate, of the system. Here, information rates of perhaps three times those of standard television systems have been considered, i.e., bandwidths of about 8 Mc. It is obvious that all components in the television system should be compatible with regard to bandwidth.

A bandwidth corresponding to the above frame rate in tube resolution is about 6½ Mc. It is expected that a slightly higher bandwidth may be employed in the surrounding circuitry of the television camera tube, so that no unnecessary degradation of signal will be introduced. Bandwidths of the order of 9 Mc have been employed in the simulation television setup for the photographs used in Vol. I of this report. However, the use of these bandwidths is not standard studio practice, because the standard-tube studio television is limited by FCC regulations to about 3½ Mc. Otto Schade, of RCA, has used bandwidths up to 20 Mc in some experimental television equipment, particularly for circuits surrounding the 4½-in. Image Orthicon camera tube.

The next component encountered by the television signal is the magnetic-tape recording system. Magnetic-tape recorders for the purpose of recording video signals have already been investigated and brought to a primitive stage of development.

A magnetic-tape recorder (Fig. 36) will be similar in many respects to the home audio-tape recorder except that it will handle much more information in a given length of time. Two reels, one for feeding the tape and one for winding it, are needed; also, the tape passes over a capstan and several other pulleys. Heads for recording information magnetically on the tape are provided, both

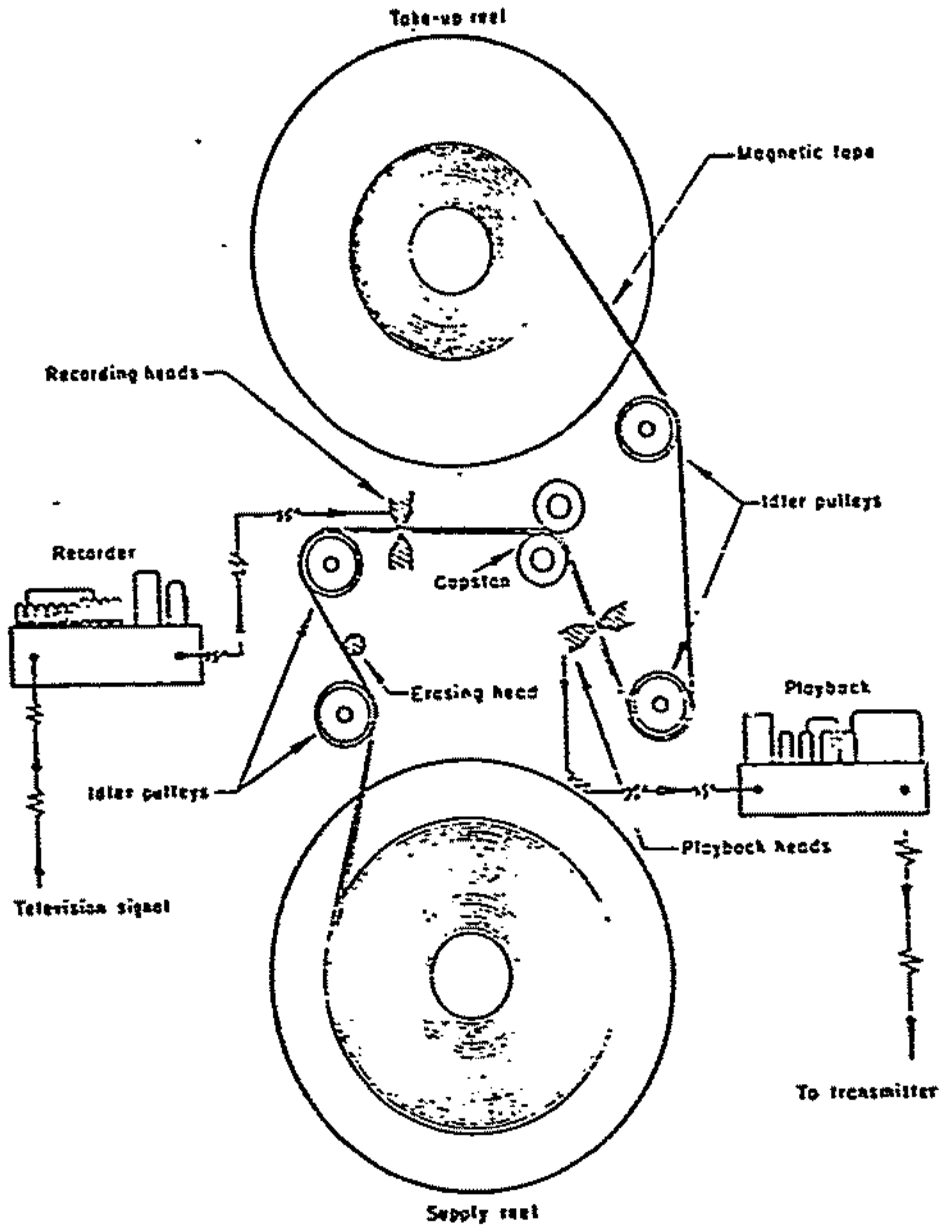


Fig. 36—Schematic of tape recorder

for recording and for playing back the information into recording heads, and for taking the information in playback.

An RCA video recording system was exhibited recently. It consists in using either a single track for the video signal, the black-and-white system, or a color system having three tracks on the tape. Tape speed is 30 ft/sec.

Bing Crosby Enterprises have a system using a somewhat slower tape speed. In their device, the black-and-white television is recorded with a number of tracks on the tape.

Both systems are designed for standard studio bandwidths and will have to be increased by a factor of two or three in order to be compatible with the bandwidth proposed for the Feed Back system. Personnel of both RCA and Bing Crosby Enterprises have expressed the opinion that within the development period allotted for Feed Back, such a recording system can be developed.

A suitable tape is one having a cellulose acetate plastic base of 0.0017-in. thickness, similar to the one developed by Minnesota Mining and Manufacturing Company. Lubricating methods developed by them are believed to be adequate. The magnetic surface of the tape is an iron oxide coating of 0.0005-in. thickness, which is impregnated on the plastic base. It is believed that the tape cannot be run continuously over the capstans for a year's period. Even if the tape itself can be made to withstand this length of service, it is probable that the magnetic heads will be worn down, because tape has characteristics not too different from those of crocus cloth. Any system assumed for the present report allows for intermittent operation and includes motors for starting and stopping the reels every time a recording or a playback is made. In fact, it is probable that the system will be started in one direction for recording and played back in the opposite direction. Discussion of the programming of the record playback magnetic-tape storage may be found under "Communication Link," page 85.

Next, in its progress through the television equipment, the signal encounters a modulator and transmitter unit. These components must have at least the 8-Mc bandwidth postulated for the other units in a television chain. Engineering of the equipment will be fairly straightforward.

The transmitter in the vehicle will be a frequency-modulated oscillator operating in the X-band and having a power output of about 10 watts. Center frequency of the transmission will be controlled by reference to a very stable high-Q resonant cavity. However, there is some difficulty in obtaining an output transmitting tube capable of transmitting at the megacycle frequency required and also having a year's life capability at reasonable power requirements.

In earlier work it was assumed that 10,000 Mc would be used for the transmitted signal. However, RCA[20] believes that 7500 Mc is a more appropriate figure, and this frequency will give a greater capability in transmission through heavy rainstorms. A frequency that is too low will require more power input to the transmitter; therefore the 7500-Mc frequency is a compromise. RCA has recommended that the output stage be a frequency-modulated magnetron with a 20-watt output for reasonably low power consumption. However, at present, magnetrons have not been developed to have a year's life, the longest, perhaps, being a month. It is probable (but not certain) that the life length of the mag-

netron can be improved. Also, it is possible that traveling-wave tubes will be developed to a state of refinement that will allow them to be considered for use as the Feed Back transmitting tubes.

For the example selected here, which was discussed informally with RCA, two klystrons developing a total of 5 watts have been used. (A larger antenna compensates for the reduction in output from the 20 watts stated above.) These tubes now have a reliability compatible with Feed Back requirements (over 10,000-hr lifetime in one reported instance). Previously, use of the klystron was not felt possible because power requirements of this tube are quite high. However, in putting together the various parts of the over-all Feed Back system, it became apparent that the 400-watt input required by the klystrons (compared with a tenth as much power required by the magnetron) was not dominant in the total payload power requirement.

Payload power requirements are already in the realm of several kilowatts, so that once a reactor is selected for the auxiliary powerplant, a ½ kw more power can be obtained for about 25 lb additional radiator weight.

A transmitting antenna with a diameter of about 5½ ft is needed for the 5-watt klystron systems. A 20-watt magnetron, on the other hand, requires only a 1-ft-diameter antenna. By placing the antennas in the locations shown on the vehicle drawing (see Fig. 1), it should be possible to enclose, within the vehicle, antennas several feet in diameter, despite their attendant complexity and weight for this particular component.

RCA has proposed an antenna system consisting of two separate paraboloid dishes: one dish receives the 5000-Mc signal and thus is able to track the ground station by means of a conical scan, and the other, the transmitting dish, is slaved to follow the receiving dish by means of a servomechanism.

Rotating parts, such as the antennas, and also the mirror wheel for the optical system, are assumed to be counterbalanced by devices of comparable moment of inertia rotating in the opposite direction.

The antenna system is to be mounted just below the throat of the second-stage rocket motor, so that upon separation of stages it will be exposed to the atmosphere and will be allowed ample freedom to scan not only directly below the vehicle, but to the horizon as well.

Approximately three video stages of amplification will be necessary between the camera equipment and the output tube. It is estimated that about 300 watts will be required to operate the transmitter circuits, exclusive of the tube requirements. The temperature of the compartment which houses the electronic equip-

ment must be regulated to within about $\pm10°C$ of a desired value, and this eliminates the need for an automatic-frequency control circuit.

A tracking command receiver will also be included in the television system. It will be a simple superheterodyne type with a bandwidth sufficient to accommodate the doppler shifts due to vehicle velocity plus an information bandwidth a few kilocycles wide, which is sufficient to permit transfer of all needed command information for the most extensive case in a period of less than 1 min.

The purpose of this receiver is to receive command information from the ground, particularly to set up the scanning, recording, playback, and transmitting operation for successive passes of the vehicle. Commands will be transmitted in the form of Baudot[23] types of symbols and will be recorded on the rotating drum of the programmer, in accordance with the present sequence arrangement, which is capable of erasing and changing all of the drum information in a period of 1 min or less. At the conclusion of each command cycle, the program drum will be played back to the ground through the data transmitter and will be checked for accuracy against the transmitted commands.

Also included in the operation is the programmer just mentioned. The programmer will probably operate in a manner very similar to that of a timer on an automatic washing machine; i.e., it will consist in a linear sequence of operations. Because successive programs differ only in variations of the length of time (including 0) that operations can take place, the required programmer is inherently simple. More comments on the ground-to-vehicle link and programming will be found under "Communication Link," page 85.

Results of RCA's investigations up to the present time are given in their various progress reports (see Refs. 13 through 21).

ENVIRONMENT PROBLEMS

Atmosphere

The physical properties of the atmosphere have been surveyed and investigated up to great altitudes. A determination of the variation in pressure, density, and temperature with altitude is based on the experimental results obtained from past rocket flights. Pressure and density values are the same as the ones presented by the Rocket Panel[26] for the region from sea level up to 250 km (156 mi). Above 80 km, the temperature is a deduced quantity and depends on certain assumptions. A temperature curve, computed at RAND,[27] falls between the maximum and minimum values proposed by the Panel for the region be-

tween 80 and 220 km. In general, the density at various altitudes is lower than the densities proposed for the "NACA standard atmosphere" in 1947. For the region from 200 km to 600 km (125 mi to 375 mi), where no direct measurements of pressure or density are available, various atmospheric models have been devised. By imposing certain requirements upon the results obtained, the model which best fulfills these requirements has been chosen to represent the state of the atmosphere at these heights. The imposed requirements have been taken from astrophysical and electromagnetic-wave-propagation studies.

At an altitude of 480 km (300 mi), the average density of the atmosphere has been found to be about 2×10^{-16} gm/cm³ and the average kinetic temperature has been found to be about 1500° ± 100°K.

In the region between 500-km and 1000-km altitude above the ground (312 mi to 625 mi), only indirect information about the state of the atmosphere is available, and these indirect data are not very accurate—i.e., as might be expected, the accuracy of the results decreases with increasing altitude. However, it has been determined that at an altitude of 500 mi the average density should be of the order of 5×10^{-17} to 1×10^{-17} gm/cm³.

Whipple[23] made an estimate of the density at these altitudes which was based on the amount of radioactive material emitted by the earth and the escape of hydrogen and helium; but this is a method which has not been used in investigations at RAND. Coincidentally, he arrives at a density of the order of 1×10^{-17} gm/cm³ at an altitude of about 500 mi.

Meteor Distribution

An estimate of the number of meteors and the amount of meteoric dust entering the earth's atmosphere has been made, based on relations between the number of observed meteors and their magnitudes, or the quantity of light they emit, when they enter the atmosphere. The amount of meteoric dust which might be present in the atmosphere has been deduced from the study of the zodiacal light, which is probably caused by the reflection of sunlight from meteoric matter.

Methods of meteor observations have been greatly improved in recent years. Bigger and better telescopes, such as the new Schmidt and super-Schmidt cameras at the Harvard Observatory and at the Dominion Observatory in Canada, permit the observation of very faint meteorites.[22] A new radar technique, which uses the ionization effect caused by the meteorities passing through the atmosphere, detects even smaller particles. By means of such direct observa-

tions, a relation can be established between the increase in the number of particles and the decrease in brightness. The increase in number of meteorites varies according to different observers, i.e., between 2.5 and 4, with decrease in brightness per magnitude. Radar observations give only information about the number of electrons produced by the particles.

None of these three methods (visual, telescopic, or radar) give information about the size or energy of the meteorites. Very little is known about the luminous efficiency of the meteorites. This latter quantity is the fraction of the kinetic energy of the original particle that is transformed into radiant energy in the atmosphere. These energies, in turn, are functions of the velocity and of the density of the particle. Calculations have been carried out for various velocities and densities which a meteorite can have. Whipple and his group at Harvard have related atmospheric densities to meteor mass magnitudes and observed velocities. Their results show that the most probable velocity of a nonsporadic meteorite is about 37 km/sec and that its average density is about 3 gm/cm³ or less.

Estimates of the numbers of meteorites which enter the earth's atmosphere every day, as obtained by different observers and by different observational methods, are shown in Table 6. The number of dust particles calculated from Van de Hulst's theory[30] of zodiacal light applies only to particles having radii between 0.025 to 0.0001 cm. Particles smaller than these will be repelled from the solar system by the radiation pressure from the sun. Numbers "observed" by radar techniques above magnitude 5 or 6 are not reliable, because radar can only detect these small particles when their velocity is perpendicular to the radar beam. It has been estimated that radar will only detect 20 per cent of the particles, if their sizes correspond to the ninth magnitude of brightness.

Energies and radii of particles having various velocities and densities are listed in Table 7 according to the stellar magnitude they are assumed to attain. In order to show the effect of velocity and density upon the number of meteors belonging to a particular magnitude and upon the corresponding energies and radii or size, calculations have been made for a velocity of 56 km/sec and a density of 5 gm/cm³, an assumption first made by Öpik[31] in 1937 and also by Watson[32] in 1941. Energies and radii have been calculated for the most probable velocity of 37.4 km/sec and a density of 3 gm/cm³, as found by the Harvard group. Furthermore, results have been obtained for particles having a velocity of 18 km/sec and a density of 3 gm/cm³, an assumption which, according to Whipple's investigation, should apply to the meteoric dust that supposedly causes the zodiacal light effect. Figure 37 contains a plot of the num-

Table 6

NUMBER OF METEORITES ENTERING THE EARTH'S ATMOSPHERE EACH DAY

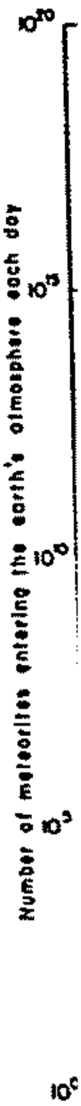
	Number of Meteorites					
Magnitude	Öpik (visual and telescopic)	Watson (visual and telescopic)	Van de Hulst* (zodiacal light)	Van de Hulst† (zodiacal light)	Millman (visual)	McKinley (radar)
−2	1.58×10^{3}	7.08×10^{4}	....	....	1.67×10^{5}	1.14×10^{5}
−1	6.31	1.78×10^{5}	....	....	4.49	2.70
0	2.51×10^{4}	4.47	....	....	1.21×10^{6}	6.41
1	1.00×10^{5}	1.12×10^{6}	....	....	3.25	1.52×10^{6}
2	3.98	2.82	....	....	8.75	3.62
3	1.58×10^{6}	7.08	....	....	2.35×10^{7}	8.61
4	6.31	1.78×10^{7}	....	....	6.34	2.05×10^{7}
5	2.51×10^{7}	4.47	....	....	1.71×10^{8}	4.86
6	1.00×10^{8}	1.12×10^{8}	4.12×10^{12}	....	4.59	1.16×10^{8}
7	3.98	2.82	6.74	....	1.24×10^{9}	2.76
8	1.58×10^{9}	7.08	1.10×10^{13}	....	3.33	6.55
9	6.31	1.78×10^{9}	1.80	....	8.98	1.55×10^{9}
10	2.51×10^{10}	4.47	2.91	....	2.41×10^{10}	3.69
11	1.00×10^{11}	1.12×10^{10}	4.80	....	6.49	8.77
12	3.98	2.82	7.85	2.01×10^{12}	1.75×10^{11}	2.08×10^{10}
13	1.58×10^{12}	7.08	1.28×10^{14}	3.28	4.69	4.95
14	6.31	1.78×10^{11}	2.09	5.37	1.26×10^{12}	1.18×10^{11}
15	2.51×10^{13}	4.47	3.42	8.77	3.39	2.80
16	1.00×10^{14}	1.12×10^{12}	5.61	1.43×10^{13}	9.17	6.65
17	3.98	2.82	9.15	2.34	2.47×10^{13}	1.58×10^{12}
18	1.58×10^{15}	7.08	1.50×10^{15}	3.83	6.64	3.76
19	6.31	1.78×10^{13}	2.45	6.26	1.79×10^{14}	8.94
20	2.51×10^{16}	4.47	3.99	1.02×10^{14}	4.81	2.12×10^{13}
21	1.00×10^{17}	1.12×10^{14}	6.53	1.67	1.30×10^{15}	5.04
22	3.98	2.82	1.07×10^{16}	2.73	3.49	1.20×10^{14}
23	1.58×10^{18}	7.08	1.75	4.46	9.39	2.85
24	6.31	1.78×10^{15}	2.85	7.30	2.53×10^{16}	6.78
25	2.51×10^{19}	4.47	....	1.19×10^{15}	6.80	1.61×10^{15}
26	1.00×10^{20}	1.12×10^{16}	....	1.95	1.83×10^{17}	3.83

*Based on $\rho = 3$ gm/cm³, $v = 16$ km/sec.
†Based on $\rho = 3$ gm/cm³, $v = 18$ km/sec.

Table 7

ENERGY AND RADIUS OF METEORITES ENTERING THE EARTH'S ATMOSPHERE

Magnitude	Energy (ergs)			Radius (cm)		
	$v = 56$ km/sec	$v = 37.4$ km/sec	$v = 18$ km/sec	$\rho = 3$ gm/cm^3 $v = 56$ km/sec	$\rho = 3$ gm/cm^3 $v = 37.4$ km/sec	$\rho = 3$ gm/cm^3 $v = 18$ km/sec
−2	6.05×10^{12}	9.06×10^{12}	1.88×10^{13}	2.64×10^{-1}	4.69×10^{-1}	9.75×10^{-1}
−1	2.41	3.61	7.49×10^{12}	1.94	3.45	7.16
0	9.59×10^{11}	1.44	2.98	1.43	2.54	5.27
1	3.82	5.72×10^{11}	1.19	1.05	1.87	3.88
2	1.52	2.28	4.73×10^{11}	7.73×10^{-2}	1.37	2.85
3	6.05×10^{10}	9.06×10^{10}	1.88	5.69	1.01	2.10
4	2.41	3.61	7.49×10^{10}	4.19	7.44×10^{-2}	1.55
5	9.59×10^{9}	1.44	2.98	3.08	5.46	1.14
6	3.82	5.72×10^{9}	1.19	2.26	4.02	8.36×10^{-2}
7	1.52	2.28	4.73×10^{9}	1.67	2.96	6.15
8	6.05×10^{8}	9.06×10^{8}	1.88	1.22	2.17	4.52
9	2.41	3.61	7.49×10^{8}	9.02×10^{-3}	1.60	3.33
10	9.59×10^{7}	1.44	2.98	6.64	1.18	2.45
11	3.82	5.72×10^{7}	1.19	4.88	8.66×10^{-3}	1.80
12	1.52	2.28	4.73×10^{7}	3.59	6.37	1.32
13	6.05×10^{6}	9.06×10^{6}	1.88	2.64	4.69	9.75×10^{-3}
14	2.41	3.61	7.49×10^{6}	1.94	3.45	7.16
15	9.59×10^{5}	1.44	2.98	1.43	2.54	5.27
16	3.82	5.72×10^{5}	1.19	1.05	1.87	3.88
17	1.52	2.28	4.73×10^{5}	7.73×10^{-4}	1.37	2.85
18	6.05×10^{4}	9.06×10^{4}	1.88	5.69	1.01	2.10
19	2.41	3.61	7.49×10^{4}	4.19	7.44×10^{-4}	1.55
20	9.59×10^{3}	1.44	2.98	3.08	5.46	1.14
21	3.82	5.72×10^{3}	1.19	2.26	4.02	8.36×10^{-4}
22	1.52	2.28	4.73×10^{3}	1.67	2.96	6.15
23	6.05×10^{2}	9.06×10^{2}	1.88	1.22	2.17	4.52
24	2.41	3.61	7.49×10^{2}	9.02×10^{-5}	1.60	3.33
25	9.59×10^{1}	1.44	2.98	6.64	1.18	2.45
26	3.82	5.72×10^{1}	1.19	4.88	8.66×10^{-5}	1.80



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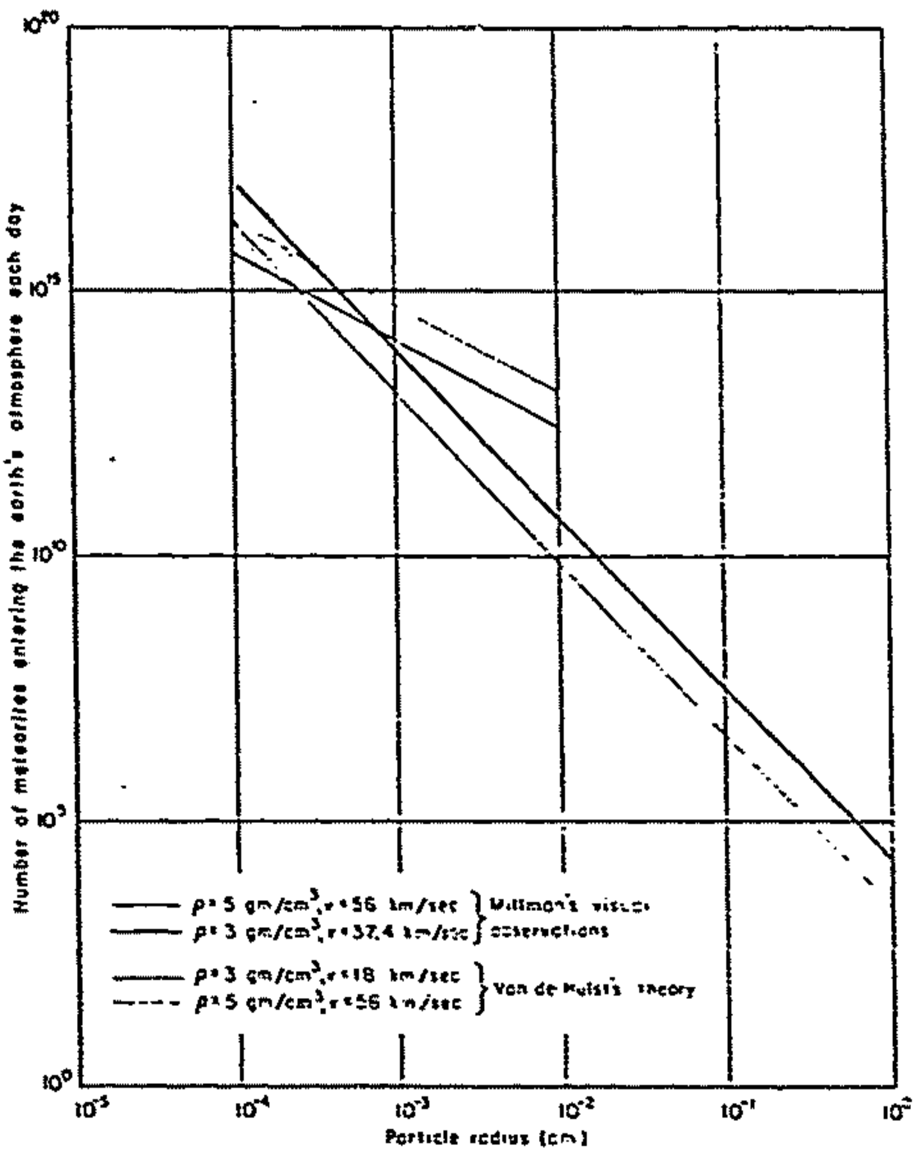

Fig. 37—Number of meteorites as a function of radii

ber of meteorites of different radii entering the earth's atmosphere at the various energies mentioned above.

It will be noted from Table 7 that the energies and radii of particles belonging to the same magnitude do increase with decreasing velocity and density. This is so because it takes much larger (slow) particles to produce the same brightness effect as small particles with high velocities. Although the number of particles having the same size increases slightly with decreasing velocity, energy is the decisive factor from the point of view of depth penetration or skin puncture, as will be shown in the following discussion.

An expected number of collisions between meteorites and the surface of the satellite has been estimated. Collision frequency as a function of meteor magnitude was calculated under the assumption that the satellite vehicle has an exposed surface area of 275 ft^2. This frequency was then coupled with penetration data to ascertain the probability of meteoric puncturing of the vehicle's skin.

Little information on the problem of high-speed armor penetration is available. To simplify the calculations for order-of-magnitude effects, it has been assumed that the shock wave set up by a fast-moving projectile entering a solid medium is presented by a right circular cone of total apex angle 60 deg. It has also been assumed that the total energy of the meteorite is used in vaporizing the material included in the volume of the right circular cone. Depth of penetration of the skin then varies as the cube root of the total energy of the meteorite. If σ is the density of the metal plate and ξ is its heat vaporization, the depth of penetration is

$$d=\left(\frac{9}{\pi\sigma\xi}\right)^{1/3}(E)^{1/3},$$

where E is the total energy of the meteorite. Calculations have been carried out for $\sigma = 7.9$ gm/cm^3 and $\xi = 1 \times 10^{11}$ ergs/gm.

Probability of hit as a function of depth of penetration is shown in Fig. 38, where the expected effects of exposing the vehicle to meteorites for periods of 1 day, 1 month, and 1 year are indicated. If the assumption is made that the skin thickness is 0.05 in., then according to this investigation, the probability that perforation occurs at least once a month is about 10^{-2}, or considerably less than one, based on Millman's observed meteor densities and a most probable meteor velocity of 37.4 km/sec.

Tables and graphs presented here refer to sporadic meteors only. The probability of hit has been calculated on the basis of Millman's visual observations, i.e., on the basis of a rate of increase of number density of 2.7 per magnitude, while an increase of 4 might be possible, as shown in Table 6.

Effects of major meteor showers have not been included in the calculation. Predictable and unpredictable showers occur at various times during the year. During the advent of a shower, which may last from a few hours to several days, the meteor rate increases considerably, and it might reach values as high as 50 to 100 per hour for a single visual observer. Under normal conditions a visual observer sees about 2 to 8 per hour. Table 8 gives a list of major meteor

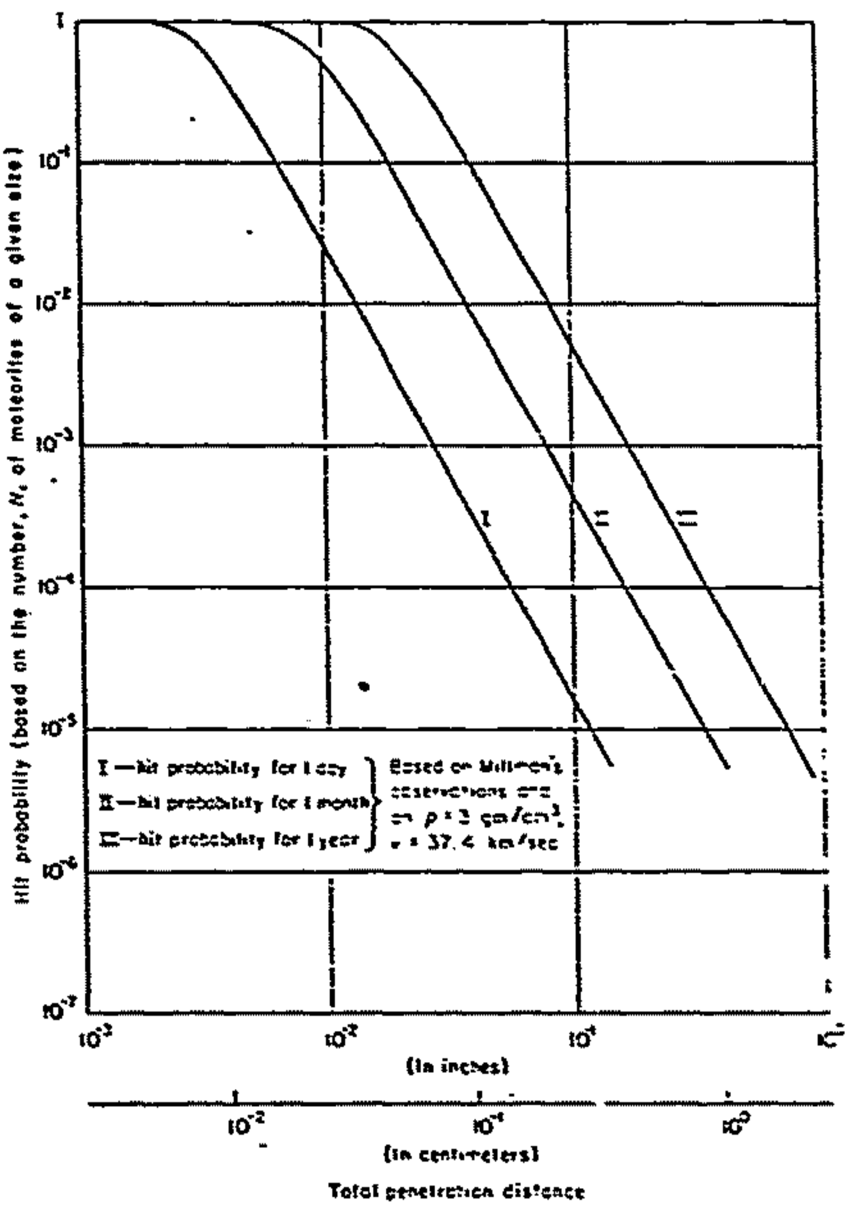

Fig. 38—Probability of penetration of a metal plate

showers which have been observed visually, showing their periods of activity, normal hourly rates, and velocities. Also included in Table 8 are the major day-time meteor showers which have been observed by the radio-wave technique. Observed showers which do not seem to occur regularly are not listed in the table.

It must be emphasized that the whole meteor problem at present is far from being well understood. Therefore, the numerical results given here must be viewed as tentative estimates.

Table 8

MAJOR METEOR SHOWERS

Shower	Date of Maximum	Normal Hourly Rate	Velocity (km/sec)
Showers Observed Visually			
Quadrantids	January 3	35	39
Lyrids	April 21	8	51
η-Aquarids	May 6	12	66
δ-Aquarids	July 28	10	50
Perseids	August 10 to 14	50	61
Orionids	October 20 to 23	15	68
Taurids	November 3 to 10	10	27
Leonids	November 16 to 17	12	72
Geminids	December 13 to 14	60	35
Ursids	December 22	13	38
Daytime Showers, Observed by Radio-wave Technique			
ζ-Perseids	June 5	40	28.8
Arietids	June 8	60	37.6
β-Taurids	July 2	30	31.5

Depth of penetration, as calculated and presented in Fig. 58 of this report, is in close agreement with the penetration depth calculated according to the theory outlined in a recent paper by Rostocker. [33]

FLIGHT MECHANICS

Only the preorbital phase of flight mechanics will be considered here. In the orbiting condition the vehicle's path can, in a sense, be considered as preordained. Perturbation from the orbit will result from errors in its establishment and from forces external to the satellite-earth two-body system. Initial-positioning error perturbation is considered under "Attitude Control," page 71, and in more detail in Ref. 34. Discussion of the effect of the variation in gravity potential on the orbit will be found in Vol. I, under "Orbital Properties of the Satellite," beginning on page 104.

A consideration of the ascent trajectory determines the value of ν (fuel/gross weight ratio), which has an important effect on the gross weight of the missile. Such a study will be found in Ref. 35. Ascent studies also contribute to the determination of aerodynamic loading and heat-transfer quantities and their effects on the value of ω (structure/gross weight ratio). Furthermore, evalua-

tion of guidance and control errors is facilitated by the use of detailed ascent computations. Results of the computations are shown in graphical form in Fig. 39 on page 71. Discussion here is confined to the assumptions and factors taken into account.

The calculations of the ascent are carried out by considering the vehicle as a mass point under the influence of thrust, drag, and gravity. The Coriolis forces arising from the rotation of the coordinate system with the earth are not included in the basic calculations because they are negligible in relation to the degree of precision required for this report. Many other small effects, such as the earth's oblateness, etc., are also ignored. However, it is necessary to correct for the rotation speed of the earth when the earth-based equations describing the ascent are transferred to the equations describing orbital motion in inertial space.

The ascent flight path can be considered to be composed of four segments: the booster stage, the satellite main-burning stage, coasting to orbital altitude, and final acceleration to achieve orbital velocity.

The equations of motion used here are essentially the same as those used in Ref. 36, except that, as mentioned above, the Coriolis force is neglected.

The vehicle is assumed to be launched vertically and, immediately after take-off, to be steered by the gimbaled rocket motors so as to place it in a zero-lift gravity-turn trajectory. The rocket motor will be able to control the vehicle's attitude in case of gusts, etc. A zero-lift path is chosen so as to yield a compromise, reasonably close to optimum, between velocity losses due to gravity and the counter-varying losses due to aerodynamic drag. In addition to its being simple to compute, the zero-lift path probably has actual merit in that it minimizes the control forces required in the ascent.

After booster separation, the main, jet-vaned motor of the satellite stage is ignited, and the vehicle continues to follow the gravity-turn path until sufficient velocity and altitude are reached. At this point the thrust is cut off and the satellite follows a power-off elliptical path until the orbital altitude is reached. When it arrives at orbital altitude, the satellite vernier motors are used to provide the velocity increment required for establishing the vehicle on its circular orbit.

No optimization of initial load factor was made for the study. The choice of an optimum load factor was determined by the interplay of drag losses, gravity losses, and burning time. Initial-load-factor values that are too low are accompanied by abnormally high gravity losses, and values that are too high are accompanied by abnormally high drag losses. In previous RAND studies, both

for the satellite and for surface-to-surface rocket vehicles, it was found that an initial thrust weight ratio of about 1.6 was a good compromise, and that ratio has been used for this report.

The performance of the satellite vehicle was calculated assuming gasoline-oxygen as the propellant for the booster stage. A sea-level specific impulse, I_0, of 249.8, which increases to a value of 282.3 above 100,000 ft was employed. For the second stage, a gasoline-oxygen propellant system having a specific impulse of 299.0 (in a vacuum) was used.

The drag coefficient of the booster and final stages was calculated, taking into account the variation of skin friction with altitude. The booster stage was assumed to be in turbulent flow all the way, since transition Reynold's numbers will be reached only during the last several seconds of flight. The satellite stage was assumed to be in laminar flow for the entire second-stage flight time. The second-stage drag curve was faired into a constant value of $C_D = 2.20$ at an altitude of about 110 mi, corresponding to free molecular flow.

To determine the ascent path with a minimum ν value, several trajectories, differing in pitch angle shortly after take-off, must be calculated. Booster paths are computed out to a time corresponding to the estimated value of ν (0.8). Second-stage calculations are then continued to a set of times corresponding to ν values slightly less than 0.8. At each of these times, the velocity and path angle for each trajectory are transferred from the rotating system to an inertial frame of reference. The velocity, altitude, and path angle in the inertial system are then used to calculate the elliptical coasting parameters of range, time to apogee, and velocity increment required at the apogee for circular orbital motion. Corrections are made for atmospheric drag losses during this coasting flight. The booster and second-stage ν values can be perturbed slightly to give equal values for each stage.

A vehicle launched from the vicinity of Fairbanks, Alaska, along the optimum zero-lift ascent path into a 300-stat-mi orbit (inclined at 83 deg to the equator and retrograde) has a mean ratio (ν) of 0.8015. The modification to this value for any other launching latitude is negligible.

The variations of velocity, path angle, altitude, and range during the optimum power-on ascent are shown in Fig. 4, page 11. The accelerations along the path due to thrust, gravity, and drag are shown in Fig. 39. The particular zero-lift ascent which has this minimum ν value is one for which the power-off coasting is about 7000 n mi. Very much larger or smaller coasting distances will require somewhat larger ν values and gross weights. For example, an

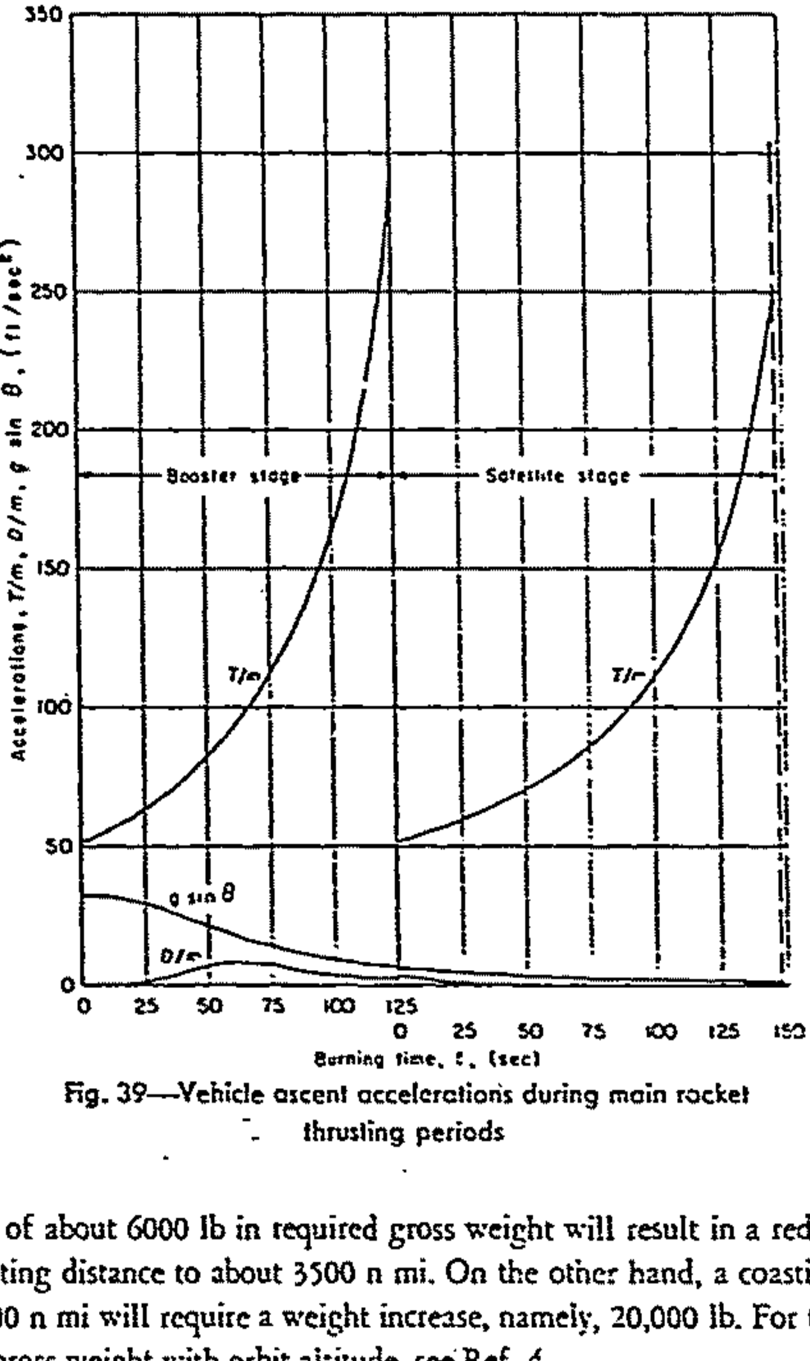

Fig. 39—Vehicle ascent accelerations during main rocket thrusting periods

increase of about 6000 lb in required gross weight will result in a reduction in the coasting distance to about 3500 n mi. On the other hand, a coasting range of 10,500 n mi will require a weight increase, namely, 20,000 lb. For the variation in gross weight with orbit altitude, see Ref. 4.

ATTITUDE CONTROL

Guidance and control of the vehicle during launching and ascent will not be described here. The guidance method assumed for this report is described in Vol. I under "Guidance and Control," page 127, and it will be seen that it does not depart materially from conventional long-range-rocket techniques.

Reference 34 contains a detailed analysis of the feasibility of a self-contained inertial guidance system.

In this section, only the control of vehicle *attitude* in the orbit will be discussed. Guidance of the vehicle per se is not pertinent, since normally the changing of the satellite path from its established stable orbit will not be considered. Once placed in its orbit with the proper velocity, the vehicle will require no further guidance, for it will pursue its course in true satellite fashion.

However, means must be provided to orient the vehicle about its center of gravity so that the television viewing system will be stabilized to the vertical, with the scanning axis lying in the orbital plane, and so that the communication antennas will be stabilized for command tracking. Stabilized orientation of the vehicle will also ensure that the auxiliary powerplant radiators will face the earth and thus be shielded from possible solar radiation and damage by meteors, although the effects of meteors are felt to be immaterial at this time.

It seems logical to describe the attitude control system in three parts, corresponding to the three main functions of the system—i.e., the perturbation torques expected, the sensing of vehicle attitude, and the control or movement of the vehicle to the correct attitude. The over-all system, and the possibilities of combining orbital and ascent control systems, will also be discussed in this section.

The information included here is a condensation of material presented in Ref. 34, a final report prepared by North American Aviation, Inc., under subcontract to RAND. North American conducted a Phase One study of attitude sensing and control for a satellite vehicle. Phase Two work, which includes some breadboard investigation of little-known phenomena encountered in connection with the previous study, has been continuing at North American, and a summary report of their findings will be available shortly.

Analysis of Perturbing Torques

The nature and size of torques which act to perturb the vehicle are presented here. Information on them is useful, since it furnishes insight into rational design of a vehicle to minimize effects of torques as well as to yield a set of specifications for the attitude control system.

A number of sources of perturbation have been considered:

1. Equivalent torques of the vehicular angular accelerations caused by orbital curvature and torsion.
2. Reaction torques from rotating parts.

3. Radiation from the vehicle.
4. Earth's gravitational field.
5. Gravitational fields of celestial bodies.
6. Earth's magnetic field.
7. Earth's electric field.
8. Atmospheric pressure.
9. Meteorite impact.
10. Light pressure.
11. Cosmic-ray bombardment.

The estimates of torque magnitudes for some effects are necessarily crude because of the present lack of exact information on the properties of the atmosphere at the orbit altitude. Moreover, the results depend to a certain extent on the vehicle configuration which is assumed. These sources will be discussed in a superficial way below. Their magnitude is given in Fig. 40.

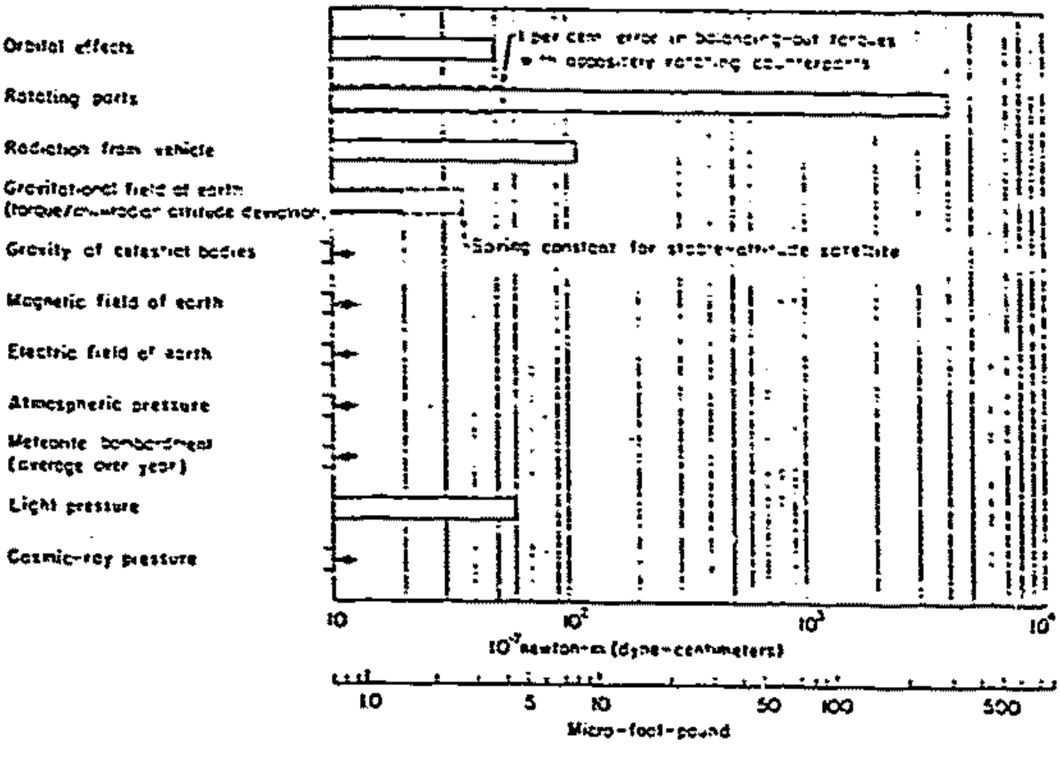

Fig. 40—Maximum estimated attitude perturbation torques

The delicacy of design required of moving components within the vehicle is made evident by the fact that light pressure is one of the more important sources of perturbing torques. Rotating parts within the vehicle can cause perturbation torques either by their acceleration or by their gyroscopic inter-

action with the pitch velocity. Acceleration is only important in a transitory sense because the torques produced are not persistent. The following example will serve to illustrate the magnitude of torque likely to result from gyroscopic interaction: Suppose that a 1-lb mass of fluid is circulated at, say, 1 ft/sec through a tube loop having a radius of 3 ft. Suppose, further, that the axis of the loop is in the pitch plane. The resulting torque is one of the order of 10^{-4} newton-m. Reference to Fig. 40 will show that a persistent torque of this value would impose considerable strain on a system designed to cope with smaller torques. Methods for handling such a situation will be discussed under "Control," page 80. Also described there will be the interaction of the stable-platform gyros with the constant pitching rotation of the vehicle.

The motion of the center of mass of the vehicle cannot directly affect the vehicle's orientation. A plane circular orbit permits the satellite to keep the desired attitude without the need for control torques; the departure of the orbit from this elementary form gives rise to a set of perturbation torques which act on the vehicle. A perturbation calculation, taking into account the ellipticity of the earth, has been made by Brouwer.[(4)] Tentative estimates of torques from this source, based on Brouwer's results, lie in the range of 1 to 5×10^{-6} newton-m (10 to 50 dyne-cm).

There are several principal sources of energy radiation from the vehicle: heat dissipation involved in the vapor cycle or in the orbital powerplant, dissipation of heat that was previously absorbed from the sun, electronic-component cooling, and microwave communication with the earth. Since the plane formed by the vehicle's longitudinal axis and zenith direction is likely to be a plane of symmetry, the center of mass and the centroid of radiation area are likely to lie on or near the longitudinal axis. Therefore, this perturbation torque will probably be one of pitch, and will probably be persistent in direction.

It can be shown that the stable attitude of the vehicle in its orbit is one in which the mass distribution of the vehicle lies generally "above" and "below" the center of mass (i.e., like a rocket vehicle with its axis vertical). This is because the mass elements below the center of mass are acted upon more strongly by gravitational forces than by centrifugal forces (which are just balanced at the center of mass). The converse is true for elements above the center of mass. Thus, any perturbation of such an attitude results in a restoring torque.

On the other hand, the conventional configuration in which the longitudinal axis of the vehicle is along the trajectory is one of unstable equilibrium and

any small attitude deviation will grow. The unbalance torque per unit angular rotation experienced for small attitude errors can be estimated by considering the vehicle to be composed of a pair of equal point masses lying (in the equilibrium case) along the trajectory. Then, if these masses have earth weight, W_e, and distance apart, l, and if the line joining them makes an angle θ with the trajectory, the unbalance torque, L, is

$$L = \frac{3}{8} \frac{l^2 a^2}{r_0^3} W_e \sin 2\theta.$$

Here a is the earth radius and r_0 is the orbital radius. For the case of a 300-mi orbit, $W_e = 1500$ lb, $l = 5$ ft; the unbalanced spring torque is 2×10^{-5} newton-m/radian.

This unbalance effect can be removed or reversed, making the desired attitude stable, by a vehicle configuration in which the lumped masses representing the vehicle lie above and below the center of mass, rather than fore and aft of the center of mass. The so-called shoe-tree configuration will accomplish this.

There are two ways in which gravitational fields of celestial bodies may affect attitude: they may affect it because of their differential attractions on the various portions of the vehicle (as exemplified by the above) or because they modify the vehicle's orbit. Insofar as they affect the orbit, they may affect attitude indirectly as explained previously. A study by Spitzer in Ref. 59 has shown that the orbital perturbations caused by the moon and the sun are overshadowed by perturbations resulting from the oblateness of the earth. Therefore, the inference can be made that the equivalent perturbation torques of these orbit changes are small compared with those already considered.

The earth's magnetic field will interact with the satellite through the permanent magnetic moments of internal components and with any fields arising from the operation of electrical equipment. Additional torques will arise from the eddy currents induced in the various conductors which comprise the vehicle. All of these effects are difficult to appraise before a complete configuration is given; further, they are amenable to design. For example, it is fairly simple to reduce the effects of eddy currents by breaking the shell of the vehicle with one or more insulated strips in order to destroy any possible large paths for current loops.

Despite the tenuous atmosphere at the expected orbital altitude (560 to 800 km), the satellite will encounter a certain air resistance, which may be characterized as that due to atmospheric pressure. In a normal attitude, with an axis of symmetry along the trajectory, the effect of this resistance will be

purely to decrease the satellite's path speed so that it will not give rise to any attitude-changing torques.

There seems to be no simple way to calculate a quantity for travel through an atmosphere of such low intensity that each molecular contact involves a collision problem. However, estimates have shown aerodynamic effects on attitude to be negligible.

As the satellite moves along its orbit, it will encounter meteorites. Those which produce a significant impulsive torque on impact are rare. The corresponding expected impulse will be of the order of 2×10^{-2} newton-m sec. An impulse of this size is significant, but since it will occur only once a year on the average, it may represent only a transient disturbance of the vehicle.

Radiation from the sun—the principal source of radiation falling on the vehicle—falls on the outer limits of the earth's atmosphere at the rate of 1.95 cal/cm^2/min. The light pressure may be considered to act uniformly over the maximum area of the vehicle projected onto a plane normal to the instantaneous line of sight to the sun. The following sample calculation will serve to illustrate the magnitude of the perturbation torque: Assume a 100-ft^2 area having a centroid displacement from the center of mass of 1½ ft, and assume that the sun is directly overhead. The torque will be equal to 6×10^{-6} newton-m. This perturbation torque can be removed entirely by changing the configuration so as to have appropriate geometric symmetries about the center of mass, e.g., by changing the configuration to a sphere or cylinder.

Data currently available[16] on the upper atmosphere indicate that the energy of cosmic-ray bombardment at the satellite may be of the order of 2 bev/cm^2/sec. This energy is small, so that even if the cosmic rays were completely reflected, their bombardment effect would be insignificant compared with that of the light rays from the sun.

Sensing of Vehicle Attitude

A number of possible ways of sensing changes in vehicle attitude have been considered by RAND[4] and by North American Aviation, Inc.[27] Of these one system appears to be quite straightforward and was selected to be used as an example in this report.

Briefly, the sensing system assumed takes on a conventional aspect inasmuch as a gyro-stabilized platform is provided to erect a vertical within the vehicle. Vertical detection is provided by an optical horizon scanner mounted on the platform (see Fig. 41). (The vertical detection is, of course, used for pitch- and

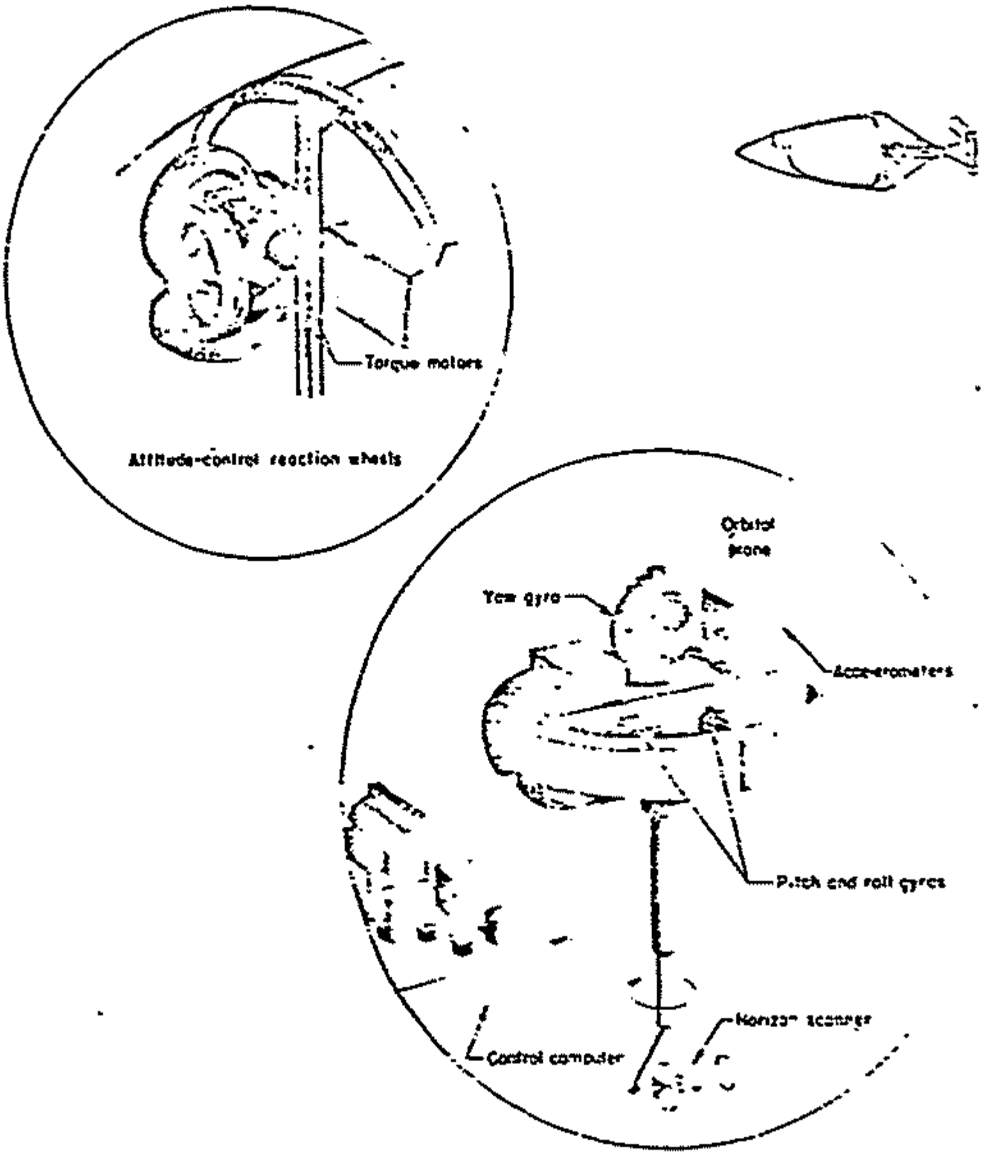

Fig. 41—Schematic of attitude control system

roll-control purposes.) Yaw sensing is provided by the azimuth gyro functioning in a fashion analogous to a gyro compass but detecting the orbital rotation instead of earth rotation, as is ordinarily the case.

Essentially the horizon scanner is a device that detects the location of the horizon of the earth. The earth is presented to the vehicle as a disk, and determination of the center of this disk yields the instantaneous vertical. The device assumed here is a telescope pointing in the general direction of the horizon and is thus at a more or less constant angle with the vertical. It is mounted on the stable platform and uses an infrared photo tube for detection.

The telescope rotates about the vertical axis and traverses the periphery of the horizon in about 1 min. At the same time, a small higher-frequency nutating scan is superimposed on this traversing motion. This is accomplished by, say, rotating a mirror at 1 rps so as to give a small angular (± 3 deg) deviation from the pointing direction of the telescope. Thus a field of view varying sinusoidally about the horizon as a datum is provided.

Considerable work has been done on horizon detection for aircraft use.[41] It is known that sufficient radiation exists at the satellite altitude (although its exact value is not known) to make a similar kind of horizon sensing system operable. In making this application the principal problem lies in the development of a sensing system of minimum weight and power consumption that will give correcting signals over a wide angle of attitude deviation.

The kind of horizon-scanning system which might be used for the sensing of the vertical is contingent upon the operating requirements demanded of it. One of the first questions to be considered is whether operation in daylight alone will suffice, or whether operation at night as well will be required.

If a conventional configuration is used for the vehicle, and a horizon scanner is permitted to cease operation during the satellite night of 50 min, then a continuously applied torque of 10^{-4} newton-m will result in the accumulation of an angular error of nearly half a radian during the unsupervised period.

There exist alternatives to continuous operation or to operation unassisted in daylight. The one assumed here is one that will maintain the vertical by means of a gyro at all times. The horizon-scanning system will be considered as a long-period monitoring system (when it receives radiation between two preset intensity levels) which will supply torque changes to gyro torquers to prevent the gyro from drifting away from its desired orientation. The supplemented gyro system resembles, in some respects, a stellar supervised inertial autonavigator system, but it is considerably less complex than the latter.

Considerations of suitable types of sensitive elements, and the justification of assuming infrared radiation, are given in Ref. 42. A number of different detectors appear feasible, including, for example, a standard 1P-21 photocell. The specific choice will be one of the more important problems involved in the design of the horizon-scanning system.

Yaw sensing is assumed to be accomplished by a gyro mounted on the stable platform and constrained so that its axis of rotation remains parallel to the horizontal as computed by the vertical detection scheme described above. Thus, the gyro axis can move only in the plane of the tangent instantaneous to the orbit path. Under such circumstances interaction of the gyro rotation with the

movement of the vehicle in its orbit path will cause the gyro axis to align itself normal to the orbit plane. Angular deviation of the vehicle heading from the orbit plane is defined as yaw.

The logic of the yaw gyro can be demonstrated by analogy with the gyro compass used in terrestrial navigation. If a gyro is placed on the earth's surface, and is gimbaled so that it can move only with its axis horizontal, a north-south orientation of the gyro axis results. Thus, the gyro senses the axis of the earth's rotation.

The gyro required for yaw detection will be one that is fairly small (weighing about a pound) and will take little power (~ 5 watts). It is believed to be reasonably determinant in development.

In case either the horizon scanner or the yaw gyro proves to be unsuitable for attitude sensing, other methods exist whereby this sensing can be accomplished.

In examining such methods, it is convenient to divide them into mutually exclusive and exhaustive classes. For example, they can be classified according to whether—

1. Observations are made entirely within the vehicle structure; or
2. Observations are made from the vehicle upon ambient field variables; or
3. Observations are made from the vehicle upon the earth or upon celestial bodies; or
4. Observations are made from the earth upon the vehicle or upon signals from the vehicle.

The second classification, for example, will require a means of sensing the only non-zero effective gravitational field at the vehicle caused by the differential radial distances of the various vehicle elements from the instantaneous center of gravitational force. Only observations of small gravitational field differences are possible, and the instrumentation problem is difficult. A small pendulum suspended by a quartz fiber is within the realm of possibility. As such, it will incorporate many of the features of instruments used in gravimetric measurements.

As criteria on the choice of sensing methods, it should be noted that the following procedures are to be avoided in general:

1. Use of a time standard, unless supplemented by a correction system.
2. Methods which require a close knowledge of the orbit prior to launching.

3. Communication of attitude information between ground and satellite.
4. Schemes requiring extensive navigational computation within the vehicle.

Control

The means of maintaining proper vehicle attitude may be defined as the muscles of the control system. The sensing units provide knowledge of the proper heading in pitch, yaw, and roll and of the angular changes which the vehicle must undergo. In general, the signals from the sensing units will originate in gimbal angle pickoffs; from the yaw gyro directly; and from the stable platform for pitch and roll. The attitude control will then be effected through a servo system registering these signals.

A number of control methods are apparent. In general, almost any effect which can cause perturbation torques can also be used to obtain control torques. Some of the control methods are

1. Reaction gyroscopes.
2. Acceleration wheels.
3. Changing the moment of inertia.
4. Jet reaction thrust.
5. Radiation from the vehicle.
6. Gravitational field of earth.
7. Magnetic field of earth.
8. Atmospheric pressure field.
9. Radiation incident on the vehicle.

The first four items are felt to be amenable to reasonable engineering treatment, although the jet reaction thrust (item 4) would probably be useful only for coarse control on occasions. Changing moment of inertia (item 3; exemplified by a falling cat righting itself) is not considered here because of complexity of design, particularly in providing control about a single axis.

A system of control based on the use of acceleration wheels (item 2), or "flywheels," has been proposed[41] and is assumed for this report. Such a system has a much simpler structure than that of the reaction gyros (item 1), since it consists of only three flywheels (Fig. 41)—their axes rigidly constrained along the three principal axes of the vehicle—and of a means for accelerating these wheels about their respective axes, i.e., torques applied by the vehicle structure to the wheels. The reaction torques of the accelerating wheels act on the vehicle frame and provide the desired control torques. The control torque impulses

which can be realized with such a system are limited only by the maximum angular momentum which can be stored in each wheel.

This method is of considerable interest, being apparently feasible, relatively simple, and able to provide a finely damped control to the vehicle.

When the flywheel system is used, a vehicle angular displacement of θ_v degrees, with a moment of inertia of the vehicle about that axis of I_v, will require a corresponding angular travel of the flywheel about that axis of $\theta_r = I_v\theta_v / I_r$, where I_r is the moment of inertia of the flywheel. Angular velocity and acceleration equations are similar, being, respectively,

$$I_r\dot{\theta}_r = I_v\dot{\theta}_v$$

and

$$I_r\ddot{\theta}_r = I_v\ddot{\theta}_v,$$

provided, in all cases, that motion about the other two principal axes of the vehicle is negligible. Since this will not be the case, the problem becomes somewhat complex, and to accommodate the cross-product terms that will ensue, a small computer will be needed.

There are two problems which must be discussed in connection with the acceleration wheel system. First, Can sufficient momentum be stored by these wheels to provide a control torque adequate to compensate all expected perturbation torques? Second, What are the control equations which should be used to determine the required control torque in terms of the sensed attitude deviation? Both are discussed in Ref. 37.

It is shown there that wheels of reasonable maximum angular velocity and size will afford sufficient momentum storage capacity to provide the required controlling torques in roll and yaw. This is partly due to the fact that there will be no persistent torques acting to perturb these attitudes, so that the average angular momentum control required over a long period of time is zero. Thus, only the total torque impulse over half of an orbital revolution need be stored at any one time. On the other hand, in pitch there will be small persistent torques, the control of which will require a constant wheel acceleration over an entire year, with a consequent excessive wheel velocity. This will be discussed next.

Reaction gyroscopes (item 1, above) of a sort will be present in the vehicle even though this system is not employed specifically for control. For example, the stable platform must be continually kept in a plane tangent to the orbital

path and thus must be continually rotated, with respect to inertial space, the rotation period being equal to that of the orbiting vehicle. A constant torque must be applied to the pitch gyro to accomplish this relative motion, and this means that an equivalent backward torque is applied to the vehicle. This latter torque is persistent, as was mentioned above, and measures must be taken to correct it.

It has been suggested that this backward torque be corrected by periodically flipping the pitch gyro back an integral number of rotations corresponding to the number of satellite orbit periods undergone in the time interval for which this correction is made. This same flipping technique can be applied not only to the pitch gyro, but to the other two gyros to account for build-up of persistent torques.

The wheel design for the flywheel system will be influenced by such important factors as

1. The adverse effect of low atmospheric pressure on heat transfer, unless the system is subjected to some environmental control.
2. Evaporation of lubricants and impregnants at low pressure and high temperatures.
3. The absence of gravitational forces, which reduces bearing loads.
4. The requirement of an operational life of 1 year.

A compromise must be made between the weight of the wheel and its spin velocity. Small wheels at high spin velocity will have increased probability of spin bearing failure, whereas heavier wheels may exceed weight and space requirements and place large loads on their supports during the trajectory acceleration phase. If the dimensional relationships of the wheels of the North American Aviation Mk-1 gyro are maintained, 8-kg wheels running at 2000 rpm will satisfy the maximum angular momentum requirements of the example.

Integration of the System

Having outlined appropriate types and characteristics of the components of the attitude control system, it is pertinent to discuss the system in general and the applicability of present techniques. As was mentioned previously, it is believed by North American Aviation personnel[37] that the roll and pitch gyros for the stable platform and the yaw sensing gyro (two units back-to-back as in the Navan gyro) can be patterned after present missile-guidance-component developments. For instance, the yaw attitude accuracy tolerance is of the order of 1 deg, and a gyro having an accuracy of 1 deg/hr will suffice.

The component requirements of the stabilized platform for ascent guidance so closely parallel those for orbital attitude control that considerable integration of the two systems appears feasible. Also, some portions of the computers in the two applications can probably be combined. During the ascent, to the end of second-stage boosting, the attitude of the vehicle will be controlled by rocket-motor forces. However, it is probable that, in the long coasting period that follows, the orbital flywheels can be employed to maintain attitude. The short burst of thrust needed just prior to orbiting will be effected by small rocket motors, which will be gimbaled for control purposes. Any excessive rotational speeds accumulated by the flywheels can be corrected at this point. Power for the guidance and control system will be provided by the auxiliary power supply during the boosting period as well as while the vehicle is on orbit.

With the above specifications in mind, it is possible to determine the approximate weights of the components that will be used. No additional allowance need be made for power supply, since the orbital auxiliary powerplant will be operating at all times.

The stable platform, including the yaw gyro, will weigh 50 lb. The main computer, which is used during ascent (and also while the vehicle is on orbit), will weigh 60 lb, and the additional smaller computer to solve control requirements arising from cross-coupling gyroscopic moments will weigh 30 lb.

The acceleration wheels will each weigh 40 lb—a total of 120 lb. The addition of 20 lb for the horizon scanner will bring the total attitude-control (and ascent guidance) weight to 280 lb.

GROUND OPERATIONS

Some of the more detailed design problems arising in conjunction with the operation of the entire Feed Back system will be described in this section. In order to support, but not duplicate, the material in Vol. I, a series of selected thoughts, suggestions, and comments are presented.

COMMUNICATION LINK

Figure 42 shows, in schematic form, the elements of the system and the communication links between them, as discussed in Vol. I. It will be sufficient to concentrate the bulk of analysis, and the direction of the over-all intelligence operation, at a center in the Zone of the Interior (ZI) and to provide a long-range communication net for contact with the outlying ground station. An air courier service will transport the bulk of the reconnaissance information from the ground station to the center.

A communications station such as that shown in Fig. 42 is designed to perform all functions of communication between the satellite and the ground and for the transmission of received information to headquarters. Transmission of data from the vehicle, and control of vehicle operation, are carried out by a system such as that diagrammed in Fig. 43. One or two ground stations are anticipated, so that periods of several hours will sometimes elapse between contacts with the vehicle. The ground facility should eventually be capable of handling several vehicles simultaneously.

Programming

Vehicle instrument readings will be handled along with the television signal; i.e., they will be stored on tape or directly transmitted, depending on the vehicle's location. These instrument readings will include such auxiliary data as gimbal angles, vehicle clock time, etc., and can be used for subsequent vehicle accessory control, as well as for proper interpretation of television information. Vehicle-control needs can be determined ahead of time and a program can be set up by command from the ground during the short part of the vehicle's flight near the ground station.

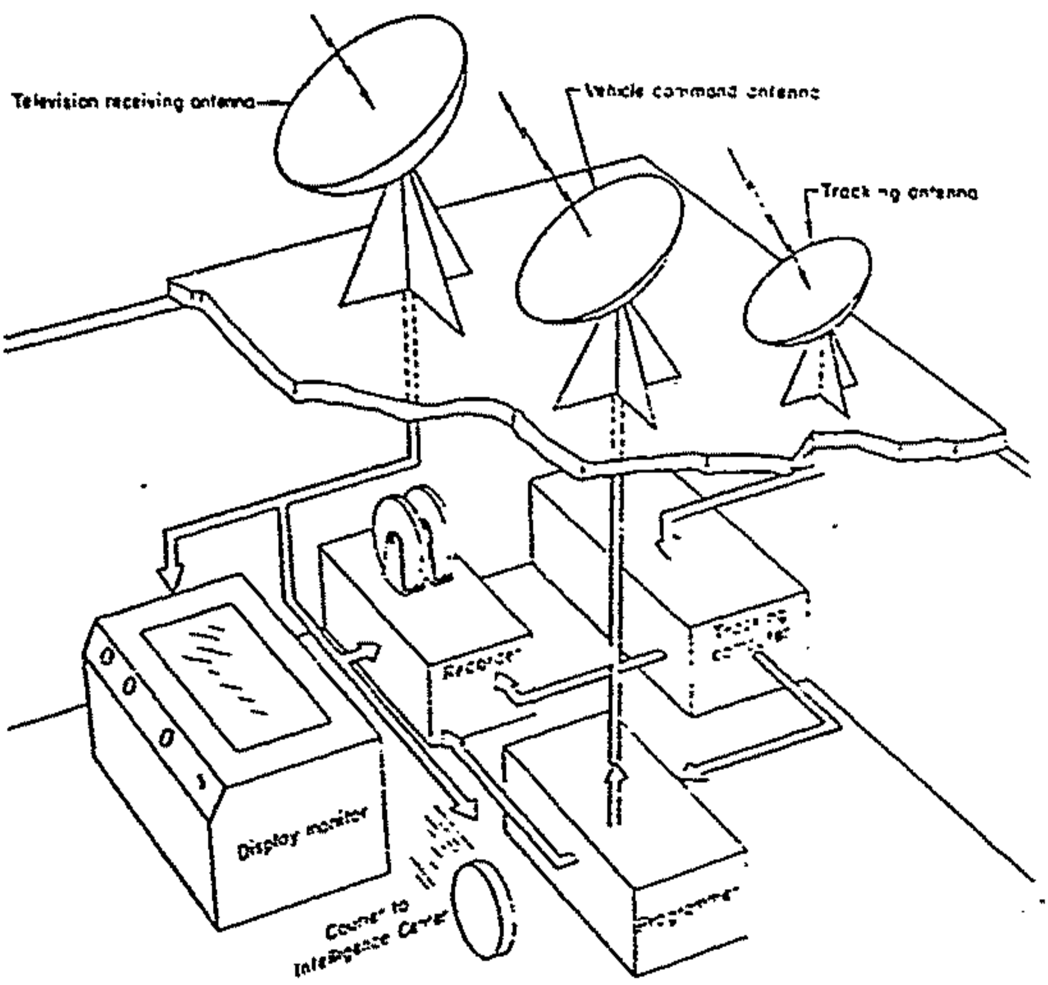

Fig. 42—Elements of the communications system

A programmer in the vehicle will consist of two parts: a clock having a short-term precision of the order of a second per day; and a memory device having a storage capacity of the order of 20,000 bits, a retention of 1 day, and an access time of the order of a second. An adequate clock can be built easily, on the basis of present-day techniques, from a tuning-fork oscillator with electronic and electromechanical counters to scale out the required time intervals. A satisfactory memory device can be made from a small loop of magnetic tape or from a small magnetic drum which rotates about once per second. Use of a magnetic unit here will give permanent retention, if necessary, as well as quick access and complete flexibility. A program established during a command cycle may also be read back and checked for correctness at the ground station in a relatively short period of time.

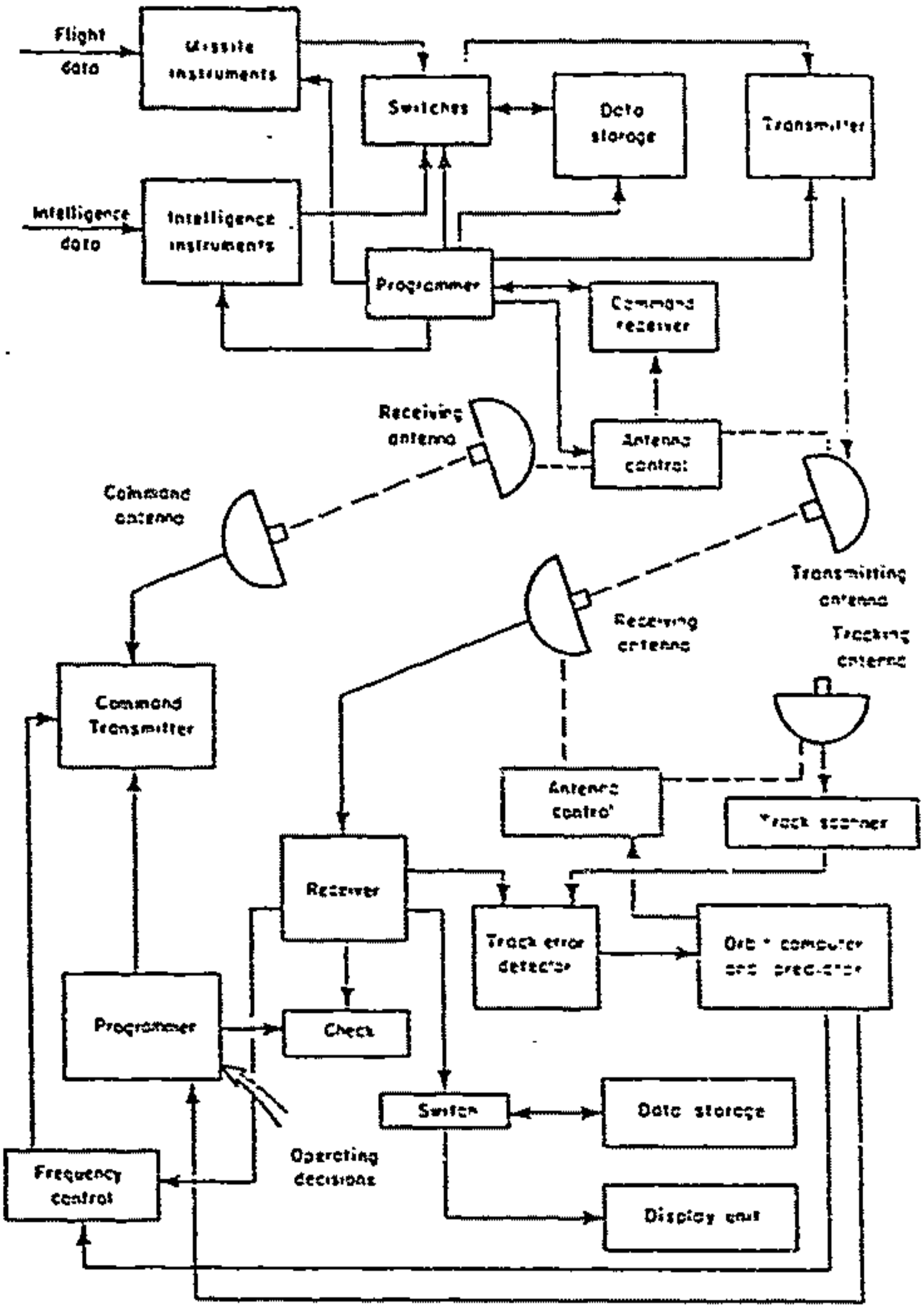

Fig. 43—Vehicle-to-ground-station communications system

Main Signal Storage

The data-storage mechanism in the television reconnaissance vehicle may limit the bandwidth performance of the whole system. A bandwidth of 10 Mc, consistent with other elements in the chain, probably can be achieved by intensive effort. The 10-Mc bandwidth will also meet most of the requirements of

electronic reconnaissance (see "Other Satellite Applications," page 99), as distinct from television reconnaissance, on a much simpler basis than would narrower bands. Magnetic tape seems to be the most logical storage medium at present, because it retains its information until erased, and yet it can be erased and reused. Multiplex systems use a number of channels on a single tape and should be compared with other possible techniques of distributing the information over the surface of the tape so as to use the resolution properties of the tape to best advantage. If each resolvable element on the ground corresponds to a resolvable element of the order of 0.001-in. diameter on the magnetic film, covering 10^7 elements per second would require only 10 in.^{2} of tape per second, against more than 400 in.^{2}/sec for the most promising present techniques. There are, therefore, some fundamental improvements possible in the field.

Transmission

In design studies to be undertaken before development for final application, the nature of the television system and the data-storage device should be considered, and the transmitter design should be integrated carefully with the requirements of these two devices. If, for example, the television and storage devices both operate on a multiplex basis, with several independent channels, it may prove practical to use several small oscillators to operate the transmitter in place of a single oscillator of larger power. Problems involved in this choice must be balanced against those of multiplex operation of the larger oscillator.

The command transmitter (ground based) can be designed to operate in the X-band with a frequency displacement from that of the vehicle transmitter which is determined by the fixed displacement in the vehicle (between the transmitter and receiver frequencies) and the doubled doppler displacement at the time of command transmission. Doppler shift can be easily computed by a parallax computer and can be used to control a variable-frequency oscillator which runs at the fixed displacement of the vehicle plus twice the doppler shift. This oscillator can then be heterodyned against the frequency of the received transmission to produce a frequency for the command transmitter. As an example, suppose that the data transmitter in the vehicle has a frequency of 9750.000 Mc when the vehicle is coming toward the ground station at 4 mi/sec radial speed; this frequency is seen at the ground station as 9750.136 Mc. If the fixed difference between the transmitter and receiver frequencies in the vehicle is 200.000 Mc, the ground computer operates an oscillator at 200.272 Mc, which, when heterodyned against the received frequency of

9750.136, gives a difference of 9549.864 Mc. This frequency, transmitted from the ground station, arrives at the vehicle as 9550.000 Mc, which is the center frequency of the command receiver in the vehicle. Such a frequency computation will help to ensure contact with the vehicle and will increase the security of the command link by providing a method of using a relatively narrow-band space-borne receiver in spite of the doppler shift.

Antenna and Control

A frequency separation of 200 Mc between transmitter and receiver, as suggested above, together with the transmitter stabilization to less than 0.1 Mc, means that waveguide filters can easily separate the frequencies and that the remainder of the components can be made wide enough to pass both with negligible difficulty. If transmitter studies show that it is possible to package the radio-frequency sections of the transmitting oscillator and or receiver on the back of the antenna system, the mechanical apparatus may become somewhat more complicated, but the radio-frequency ducting will be considerably simplified.

On receipt of orders from the programmer, antenna controls must position the reflector to an accuracy of about 1 deg with respect to the vehicle frame. Vehicle antenna motions probably should be slow rather than abrupt, and the angular momentum of the rotating antenna should be balanced by the angular momentum of the control motor rotating in the opposite direction, so that the net reaction on the vehicle control system will be kept small.

Maximum angular tracking rates of the ground antenna, if patterned after radar mounts, will be 1 deg/sec or less. It is therefore suggested that the fixed axis of the tracker be horizontal, and that a movable axis at right angles to the fixed axis carry the large reflector. Mechanical problems of a large reflector of 20-ft diameter or so are not great. For example, the tolerances on the 50-ft-diameter reflector at the Naval Research Laboratory are considerably tighter than those on the antenna assembly proposed here. A tracker can be housed in a spherical rubberized-cloth radome similar to that used on the AN/CPS-6B radar set, so that no serious climatization problems will occur even in arctic operation. Radio-frequency components of the ground transmitter and receivers can be mounted on the back of the reflector, as is done in the space craft. The antenna control motors will move the antenna in its coordinate system in response to a program set up by the orbit predictor and corrected by the track-error detector.

Prediction of the orbit for a day or several days in advance may be required for operational planning. To accomplish such prediction, information on the existing orbit is required for control of the antenna; information on the past motions of the vehicle is required for data interpretation. These needs suggest that the orbit computer and predictor contain two storage units for the accommodation of present and future orbit information, and that records resulting from the correction of the present orbit record based on tracking information be prepared for use in the data presentation and display units. Of the current methods of storage, probably the most satisfactory is the digital printing of numbers on magnetic tape. An orbit computer is required to evaluate the parameters of the orbit and to use these parameters to predict the future orbit. It must also compute parallax between the orbit and the station position and between the orbit and any other selected point on the earth as a function of time in order to control the tracking antenna and to provide interpretation data.

A programming unit in the ground station is the point at which operating decisions are put into the system. It includes a presentation device that shows the locations of the vehicle being directed at future times of interest, together with its path and estimated coverage pattern; a keyboard for formulating instructions for the vehicle; a storage device to hold the instructions for later transmission; and a device that is used to check the prepared instructions, as well as the instructions which are repeated back from the vehicle, for possible errors, since failure of this operation may cause loss of future contact with the vehicle and result in failure of the mission, or at least necessitate a separate search and recovery procedure.

The frequency control and command transmitter required for transmission of orders to the vehicle takes computed doppler frequency from the orbit computer, vehicle transmitter frequency from the receiver, and a fixed-frequency difference from a built-in crystal oscillator and generates a command signal, on the correct frequency, which will enter the pass-band of the command receiver in the vehicle. Command transmission involves a relatively small amount of information; therefore, the bandwidth need not be great, and the modulation method is chosen for reliability and freedom from noise or interference under actual conditions, rather than for maximum-information capacity of the circuit under ideal conditions.

Display for television will conceivably involve three classes of equipment: Immediate inspection of returned data, in order to derive weather information,

will require the use of a photographic mosaic device having a large sheet of film and operating in parallel with the detail data storage to mosaic the returned information on a scale convenient for cloud inspection. Film from this device can be developed rapidly after each pass of the vehicle, and the resulting strip mosaics can be laid on a map of proper scale to provide a daily mosaic containing all of the cloud pictures available. This mosaic will be inspected by weather specialists at the ground station, and the resulting weather information will be sent by radio, arriving several hours in advance of the air courier.

B-1

LAUNCHING CONSIDERATIONS

Some of the additional launching considerations too detailed for inclusion in Vol. I will be given here. Figure 44 shows the estimated time scale of the various launching operations, many of which have been described previously and a few of which are similar to procedures encountered in a conventional rocket firing. The total of 4 hr is based on a trouble-free take-off. As is customary, the count-down will be held stationary while any unforeseen adjustments are made. Launching should take place within about an hour of the scheduled time, since the orbit plane will be affected. If this cannot be done, then launching should be held over until the following day, at the same solar time. A series of such one-day delays can be absorbed without appreciable effect on the final result.

In general, equipment within the vehicle will be preheated to anticipated

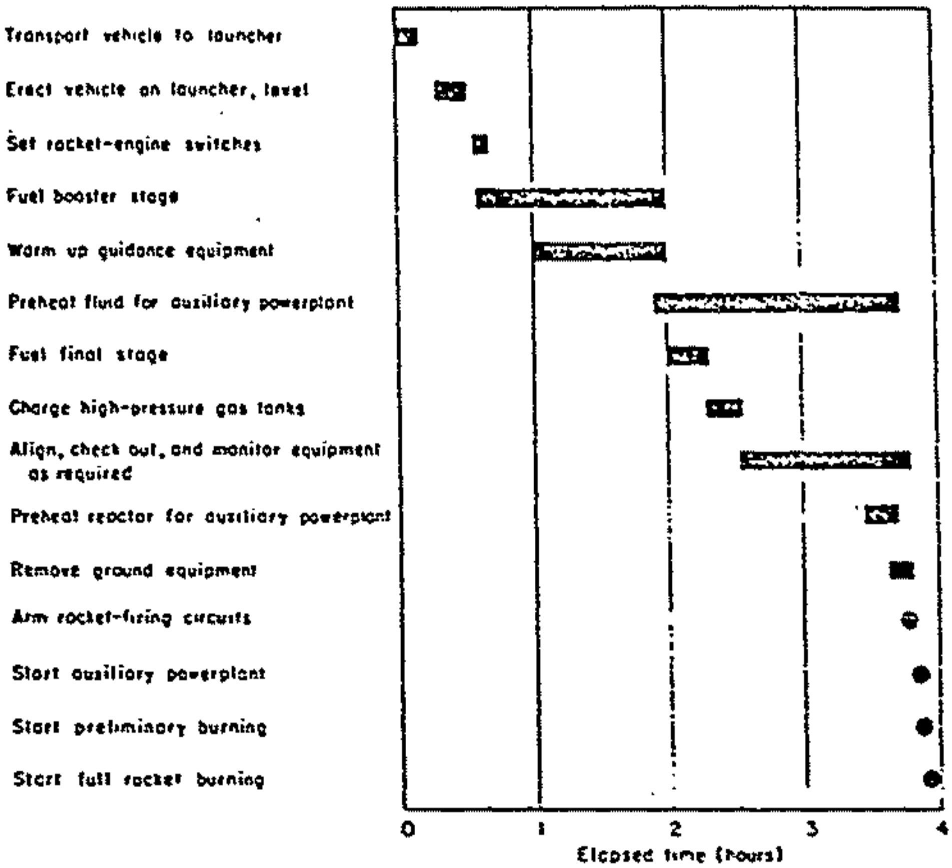

Fig. 44—Preliminary estimate of Feed Back launching operations

orbital operating conditions and will be maintained thus for a sufficient length of time to allow temperatures to approach equilibrium. After preheating, the system will be monitored for steady-state behavior before launching. Guidance-component alignment will be completed by setting the time program at zero and checking all system outputs for proper initial values. The azimuth gyro can be aligned and monitored by observing a mirror through an observation port in the final stage by means of a ground-mounted optical fixture. To achieve the desired accuracy in azimuth (about 0.1 deg), this monitoring and alignment should continue until at least 15 min before launching.

While the vehicle is on the launcher, the guidance-compartment temperature tolerance is not severe, being of the order of 20° to 30°F; however, some small components will be kept within 1°F of a specified temperature by means of vacuum insulation. All electronic components will be encased in evacuated containers to approximate design conditions. During the prelaunch checkout, provision must be made to cool operating electronic components by means of an

external unit mounted on the workstand. The orbital cooling system for the guidance compartment must also be checked before the vehicle is launched. It may be desirable to pack dry ice, or some other expendable coolant, around equipment subjected to heat during operation of the rocket powerplants.

Ground Operations—Rocket Powerplants

Since the first- and second-stage rocket powerplants will have been thoroughly tested prior to installation in their respective sections and shipment to the launching base, there is no requirement for static test facilities at the launching base. Each motor will be tested at full thrust at facilities in the ZI, where it will be cleaned, purged, treated for corrosion prevention, and installed in the proper airframe section, which will, in turn, be packaged for shipment. Each propulsion system will be given a cold run (using water instead of propellants) at the launching base as a final check for leaks and for the functioning of valves, injectors, and cooling systems.

On the launcher, the booster rockets will be armed and started in a manner similar to present-day rockets, involving a sequence of events such as the following: After fueling is completed, links are installed in each thrust chamber; these links rupture when the igniters are burning properly. Starting is accomplished in two stages—preliminary burning and full burning—by an automatic sequencing control system. First, the propellant tank vent valves are closed, the tanks are pressurized, and propellants are valved to the igniters, which begin to burn, rupturing the igniter links. The main propellant valves then open partially to allow preliminary burning to begin. After proper combustion is obtained, a second automatic sequence begins, which starts the gas generator to supply hot gas to the turbines. When pump pressures reach a preset value, the main propellant valves open to allow full burning to begin, and the launching is completed.

Propellant Storage and Handling

Approximately 11,000 gal of liquid oxygen (LOX) and 7500 gal of special gasoline will be required for each vehicle. Since the launching rate will probably not exceed one or two vehicles per month, propellant storage requirements will be minimum. One standard railway tank car can store enough gasoline of the type required for one vehicle. Enough LOX for one vehicle can be stored for as long as a month in two special railway tank cars equipped with vacuum-insulated LOX tanks having a loss rate of less than 0.5 per cent per day. To

reduce propellant handling and attendant losses it may be desirable to fuel the vehicle directly from these tank cars. An auxiliary LOX trailer of approximately 500-gal capacity will be provided for the topping of the vehicle LOX tanks shortly before launching. The usual precautions for handling these propellants will be observed, including protective clothing and the elimination of spark and contamination hazards.

Table 9 lists the general type of equipment required to assemble, test, transport, and launch satellite vehicles. Quantities are not given, since they depend on the launching rate; however, one item of each type listed, plus spare parts, will probably suffice for launching as often as once per week. Government- and contractor-furnished items are not differentiated.

Table 9

EQUIPMENT REQUIRED TO ASSEMBLE, TEST, TRANSPORT, AND LAUNCH SATELLITE VEHICLES

ASSEMBLY EQUIPMENT

1. Assembly dollies for the three main sections (satellite, booster tank, booster propulsion)
2. Beams and slings for hoisting main sections and assembled vehicle
3. Crane, overhead or mobile, for hoisting main sections or assembled vehicle
4. Test equipment for satellite communications and reconnaissance system, including
   a. Television camera
   b. Optical scanner
   c. Antenna and control
   d. Command receiver
   e. Transmitter and modulator
   f. Programmer
   g. Data storage and playback
   h. Switches and relays
5. Test equipment for auxiliary powerplant, including
   a. Heat cell
   b. Radiator, condenser, boiler
   c. Pumps and valves
   d. Engine and generators
6. Test equipment for guidance system, including
   a. Ascent control system
   b. Orbital attitude control system
   c. Horizon sensing system
7. Test equipment for propellant feed system
8. Unit—propellant tank and feed system purging and pressurizing
*9. Unit—electronics ground cooling
*10. Unit—auxiliary powerplant preheating and circulating
*11. Generator set—external power supply
12. Umbilical set—external power supply

*These items can be portable for use in the prelaunch checkout.

TABLE 9—continued

TRANSPORTING EQUIPMENT

1. Trailer—satellite transporting and erecting
2. Truck-tractor—satellite prime mover

LAUNCHING EQUIPMENT

1. Platform—launching
2. Workstand—prelaunch, truck mounted
3. Test equipment—prelaunch monitoring
4. Panel—rocket starting and launching
5. Unit—auxiliary powerplant preheating and circulating
6. Unit—electronics ground cooling
7. Generator set—external power supply
8. Umbilical set—external power supply and launching control
9. Fixture—azimuth gyro alignment
10. Trailer—main liquid oxygen servicing and storage (or tank car)
11. Trailer—auxiliary liquid oxygen servicing
12. Trailer—gasoline servicing and storage (or tank car)
13. Trailer—high-pressure gas servicing and storage
14. Vacuum pump—diffusion type
15. Ground tracking equipment—optical
16. Ground communications equipment

ALASKA AS A LOCATION FOR FEED BACK FACILITIES

One of the interesting locations for both the launching and the communication facilities is the north-central region of Alaska. Some of the factors involved in the construction and operation of these facilities are discussed below. Fairbanks and Point Barrow have been chosen as representative locations in this area, since they differ markedly in regard to accessibility and development.[11]

Transportation in Alaska

Transportation methods in Alaska may be listed in the order of their importance: air, coastal and inland water, highway, and railroad. Air facilities are plentiful, and lake and river areas serve as landing bases for float-type aircraft in the summer and for ski-equipped craft in the winter. There are numerous ports suitable for docking ocean-going cargo vessels, but most of these are closed by ice from November to April. The more northern ports and landings, such as Point Barrow, are usually open to boats from July to September only. Inland waterways afford access to central and east-central Alaska, by means of river steamers, during 5 months of the year. The Alaska highway

connects Fairbanks with Canada and is kept open all year. Although there is an improvement program under way, this highway, as of 1950, was mostly gravel or crushed rock and was very rough during the winter. A typical secondary highway runs 162 mi from Fairbanks to Circle and is a gravel road 10 to 18 ft wide. This road is closed from November to May except for the first 30 mi; it is very dusty (or muddy) and has steep grades and sharp turns which make it hazardous to traffic even when it is open. The only standard-gauge railroad in Alaska runs 470 mi from Seward to Fairbanks over rugged terrain. This railroad is kept open all year and is equivalent to a second-class line in the United States. It is clear, then, that air transportation is the only method which reaches all areas of Alaska at all times of the year.

Communications

The Alaska Communication System, operated by the Signal Corps, provides the only landline telephone and teletype channels of any consequence, running from Seattle to Fairbanks. Areas to the north and east of Fairbanks rely on radio for all long-distance communications.

Weather

Average year-round weather at three locations in Alaska is rated from the standpoint of operating conditions in Table 10. There is no significant difference in over-all weather at these locations, widespread as they are; therefore,

Table 10

OPERATIONAL RATING OF WEATHER AT THREE ALASKAN LOCATIONS

(1 = best, 3 = worst, for Feed Back operations)

	Anchorage	Fairbanks	Point Barrow
Mean minimum temperature	1	2	3
Absolute minimum temperature	1	3	2
Mean precipitation	3	2	1
Mean number of days with precipitation	3	2	1
Mean number of days with snowfall	3	2	1
Mean number of days with 6-in. snow	1	2	3
Surface winds (none extreme)	2	1	3
Mean number of clear days	3	2	1
Mean number of days with visibility less than 2 mi	1	2	3
Mean number of days with morning fog	1	2	3
Mean number of days with ceiling <1000 ft and visibility <2½ mi	1	2	3
Thunderstorm frequency	2	3	1

the choice of launching and communications sites will probably be made on some basis other than weather.

Point Barrow has the following characteristics as a launching or communications site:

1. Minimum hazard from launching test or operational vehicles on headings >260 deg or <90 deg.
2. High latitude good for maximum communication with vehicle.
3. During winter, is accessible by air only; present landing facilities would have to be improved for aircraft larger than C-54.
4. No landline communications at present.

Fairbanks has the following characteristics as a launching or communications site:

1. Rail and highway connections, although air transport would probably be preferred.
2. Well-developed air base facilities.
3. Landline telephone and teletype connections with the United States.
4. Launching NNW (as for an 83 deg orbit) would be over very sparsely populated area, although Fairbanks is inferior to Point Barrow in this respect.

Construction

Scheduling of base construction in Alaska should take into account the weather and transportation factors which limit the available construction time during the year. At Point Barrow, for example, excavating and earthwork are feasible for only 2 to 3 months. If construction materials are delivered by boat to Point Barrow, only a 3-month working period remains before winter sets in. This may be extended to 5 or 6 months if materials are delivered earlier by air. In any event, if planning, procurement, and shipment are carefully timed, and if prefabricated structures are used wherever possible, the construction of communications and launching facilities can be completed within 1 year. Previous experience has shown[44] that Loran and radar installations can be built and operated along the northern coast of Alaska, and it is believed that the problems of construction and operation of Feed Back communications stations will be similar to those experienced in building and operating the Loran and radar installations.

OTHER SATELLITE APPLICATIONS

Quite a few possibilities present themselves for satellite application. Among the more obvious, and one seriously discussed by various Air Force scientific personnel, is the use of the satellite to conduct medical and biological experiments in the fields of no gravitation and intense cosmic rays.

B-1
thru
p 102

A second type of instrumentation is one employing a 10-in.-or-larger telescope. This will allow detailed observation of solar meteorology and will also give data on the atmospheric composition of other planets, particularly Mars and Venus. Outside the solar system, useful data can be obtained on such important phenomena as the structure of stellar atmospheres, the color temperature of hot stars, eclipsing binary stars, absolute star magnitudes and distances, and the nature of supernovae. Such a telescope will increase very basically our understanding of what goes on in the stars, and in the spaces between them, and thus will add to our fundamental physical knowledge of the behavior of matter under conditions not attainable in the laboratory.

A third type of instrumentation is that of a reflecting telescope having a still larger aperture. Such an instrument will have great advantages over present ground-located instruments whose "seeing" is greatly degraded by the atmosphere. A large space-borne instrument, while many years in the future, will yield information on the extent of the universe, on the structure of galaxies and globular clusters, and on conditions on other planets of our solar system. Such a new and powerful astronomical tool may do more than supplement our present conceptions of the universe we live in; it can conceivably uncover new phenomena not yet imagined and perhaps modify our basic ideas of space and time, of energy and matter.

The Satellite as an Upper Atmosphere Station

Knowledge of the earth's upper atmosphere is important in predicting weather, in establishing reliable communication, and in developing high-speed aircraft, rockets, and guided missiles. This knowledge is at present very meager and rests on a few direct observations made from short flights of sounding rockets and on much indirect evidence and conflicting theory. A satellite may obtain data, continuous in time and covering wide geographical areas, which can significantly advance our understanding of the earth and its atmosphere. Spectroscopic observations of various atmospheric layers, obtained at glancing incidence, will indicate the electrical and chemical composition of the atmosphere. Radio reflections from the ionosphere will yield more precise information concerning the height and composition of the F_2 layer, as well as its normal diurnal motion and its abnormal motion during magnetic storms. Direct measurements of the earth's electric and magnetic fields, their geographical variation, and their fluctuation in time will be of fundamental value in helping us to understand the wind motion of the upper atmosphere and the accom-

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panying complex magneto-hydrodynamic effects. Data on the earth's magnetic field during solar disturbances will assist us in understanding the influx of charged cosmic-ray particles. They will also be of potentially great practical value in predicting the extent and duration of magnetic storms which black out radio communication. Finally, the satellite will provide, for the first time, a complete picture of the dynamic weather pattern over land and sea, thus adding a fundamental tool to the science of meteorology. The direct transmission of high-frequency radio to and from the satellite should be a very valuable source of information on the effects of the atmosphere on such wavelengths, particularly when rays of low elevation with the ground are employed.

Meteorites can be included in the category of the upper atmosphere, and the satellite can be employed to obtain data on their distribution, frequency, and impingement effects. By-products of these data will be a better knowledge of the effects of very-high-speed penetration of particles and of some of the physical phenomena surrounding such a process, e.g., the determination of the equation of state for materials.

The Satellite as a Cosmic-ray Laboratory

Cosmic rays are of fundamental importance in nuclear physics and cosmology. Some of the high-energy particles bombarding the earth constantly are much more energetic than any that man can produce. They provide the most powerful probe known for investigating the properties of atomic nuclei. Research on the origin of cosmic rays and their source of energy will lead to a better understanding of the magnetic fields of the sun and the galaxy and may shed light on the age, size, structure, and evolution of the universe.

A satellite cosmic-ray station can measure the directional intensity of primary cosmic radiation (protons, alpha particles, and some heavy nuclei). Absorption measurements will help to determine the energy and composition of the incoming particles and will be free from the complicating effects of the secondary cosmic rays produced in the atmosphere. Ultimately, observations using cloud chambers and photographic emulsions will be feasible, and these will provide a better understanding of the energy distribution of the primaries and the phenomena of high-energy-particle physics.

APPENDIX

CONSTANTS OF SUGGESTED OPTICAL TELEVISION SYSTEMS

(Altitude, 300 mi; velocity corrected for 59 deg latitude and ground speed)

	Mapping System
Scale factor	500,000/1
Pickup tube (Image Orthicon)	RCA-5826
Aspect ratio	4:3
Photocathode image size	0.96 × 1.28 in.
Scanned area on ground	7.57 × 10.1 mi
Scanning angle for one ground area	1.76 deg
Angle of diagonal of unit area	2.35 deg
Total scanning angle	62.7 deg
Total scanning width	374. mi
Picture frame rate	23.1/sec
Picture frame time	0.0432 sec
Scanning time per strip	1.56 sec
Forward (vertical) distance traveled per strip	6.82 mi
Frequency band	6.52 Mc
Distance corresponding to one line	71.7 ft
Light intensity at ground scene	10,000 ft-c
Total lines per frame	600
Active lines per frame	558
Vertical blanking	7%
Vertical blanking period	0.00302 sec
Scanning line frequency	13.87 kc
Optical pulse time	0.00155 sec
Pulse immobilization	0.5 line
Necessary image scanning	0.0
Percentage unimmobilized	0.1%
Optical system focal length	38 in.
Speed (no filter)	f/24

Mapping System

Speed (with filter, K2)	f/17
Field diagonal	2.35 deg
Horizontal response*	(f/24) 80%†
Horizontal response (filter)*	(f/17) 84%†
Vertical response*	(f/24) 85%‡
Vertical response (filter)*	(f/17) 89%‡
Drum diameter	(f/24) 26 in.**
Drum diameter (filter)	(f/17) 36.5 in.††
Rpm (drum)	38.5
Segments in strip	36
Mirror pairs	18

*16 TV lines/mm.
†At 24.6 TV lines/mm.
‡At 15.5 TV lines/mm.
**$R = 8.04 \times$ aperture.
††$R = 1.89 \times$ aperture.

B-1

MAPPING SYSTEM

Scale factor: 500,000.
Altitude: 500 mi.
Aspect ratio: 4/3 (the larger dimension measured perpendicular to the direction of motion of the satellite).
Image Orthicon photocathode image size: 0.96×1.28 in.
Scanned area on ground directly beneath satellite: 7.57×10.1 mi.
Angular velocity of satellite: $15.3 \times$ equatorial angular velocity of earth.
Equatorial diameter of earth: 7927 mi.
Equatorial circumference of earth: 24,900 mi.
Angle between orbital plane and polar axis: 7 deg.
Overlap ratio of adjacent ground areas:

$$O_H = \frac{\Delta H}{H} = 0.1\,,$$

$$O_V = \frac{\Delta V}{V} = 0.1\,.$$

This amount of overlap is necessary because of the possible variation in altitude. At approximately 59° N. latitude, the ground speed of the satellite is $15.186 \times$ the equatorial velocity of the earth's surface. Then the projected ground speed of the satellite is

$$V_{so} = \frac{(24{,}900)(15.186)}{(24)(3600)}$$
$$= 4.375 \text{ mi/sec}$$
$$= 23{,}100 \text{ ft/sec.}$$

The scanning angle for an area 10.1 mi wide is 1.93 deg. For $\Delta H/H = 0.1$ and equal angle viewing, each scanned area will progress out from center by increments of 1.74 deg. If we use an optical system having 18 mirror pairs (giving 36 scanned areas across the strip),

$$\text{Total scanning angle} = 35(1.74) + 1.93 = 62.7 \text{ deg.}$$

Taking into account the curvature of the earth's surface, the

$$\text{Scanned strip width} = 374 \text{ mi.}$$

If the earth's surface is assumed to be flat across the strip, the

$$\text{Scanned strip width} = 365 \text{ mi,}$$
$$\text{Error} = 2.4\%.$$

Time required for the satellite to travel 0.9 of the forward dimension of the scanned area is

$$T = \frac{0.9 \times 7.57}{4.375} = 1.557 \text{ sec,}$$

or 36 scanned areas across the strip.

Frame time is

$$T_F = \frac{1.557}{36} = 0.0432 \text{ sec.}$$

Frame rate is

$$F_F = \frac{1}{0.0432} = 23.1 \text{ cps.}$$

The number of active scanning lines is chosen to be 558 in order to give a reasonable approximation to "flat field scanning":

Total lines = 600,

Vertical* blanking = 7%,

Active lines = 558,

Vertical blanking interval = 0.00302 sec.

*The terms horizontal and vertical are used here in the same sense as in present commercial television practice; i.e., *horizontal* is the direction parallel to the scanning lines, and *vertical* is the direction normal to the scanning lines.

Scanning-line frequency and duration is

$$f_H = (600)(23.1) = 13.87 \text{ kc},$$
$$T_H = 72.1\ \mu s,$$
$$\text{Horizontal* blanking} = 15\%,$$
$$\text{Active trace time} = 61.3\ \mu s.$$

Count-down ratios from 13.87 kc to 23.1 cps:

$$4, 5, 5, 6.$$

Width of one scanning line represented on the ground is

$$W' = \frac{1}{558}(40{,}000) = 71.7 \text{ ft}.$$

Isolated objects of this dimension and having sufficient contrast will be detected. If there is a configuration of objects, the results depend, to a large extent, on the configuration.

Image immobilization: To restrict the movement of the image to a distance of ½ the width of a scanning line, the motion shall be no more than 35.9 ft on the ground. Then the

$$\text{Time to travel this distance} = \frac{35.9}{23{,}100} = 0.001555 \text{ sec}.$$

Using the technique of pulsing the photocathode of the Image Orthicon, the duration of the pulse will then be 1.55 ms. For example, assume the limiting horizontal resolution in the television system is to be 600 lines; the fundamental frequency for this number of lines is

$$\begin{aligned} f_n &= A_r n \frac{10^6}{2Ha} \\ &= \left(\frac{4}{3}\right)(600)\left(\frac{10^5}{2 \times 6.13}\right) \\ &= 6.52 \times 10^6 \\ &= 6.52 \text{ mc}. \end{aligned}$$

The required video bandwidth is then 6.52 mc.

B-1
thru
p 110